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VOLUME III: C-5A HEAVY LOGISTICS TRANSPORT VALIDATION

LOCKHEED-GEORGIA COMPANY
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This study was conducted to validate military specification MIL-F-9490D, "Flight Control Systems-Design, Installation and Test of, Piloted Aircraft, General Specification for," dated 6 June 1975 by checking the specification requirements utilizing the experience and knowledge derived during the procurement of the C-5A Heavy Logistics Transport.

This validation was based on test and design requirements, existing ground

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test, flight test, "in service" usage, and analytical data as was available for this validation process.

Each applicable paragraph was examined with regard to practicability, accuracy, and completeness as a requirement for procurement, design, test and installation of flight control systems for future piloted military aircraft.

Recommendations have been made with regard to changes considered necessary to improve the practicability, accuracy, and completeness of the specification and to improve or update the Users' Guide.

LEVEL II

SUMMARY

This report represents the results of a study to validate Military Specification MIL-F-9490D (USAF), "Flight Control Systems - Design, Installation and Test of Piloted Aircraft, General Specifications for," dated 6 June 1975, by performing a detail comparison of its requirements with the C-5A development and characteristics.

The comparison was based on the initial C-5A and the Active Lift Distribution Control System design, development testing and analytical requirements and results. Test and analytical data are presented or discussed where appropriate. If the requirement was not met, the reasons why compliance was either not necessary or not desirable were usually given. When appropriate, an assessment was made as to whether the requirement stringency is too lenient, good or too strict. If a change was believed to be necessary to improve the practicability, accuracy or completeness of the requirement, a recommendation for change is made and supported.

Lockheed concluded that some changes were desirable; that the specification represents a worthwhile advancement towards clarifying flight control system related procurement requirements; and that, with recommended revisions, it is applicable to future transport type aircraft.

Problems were experienced in interpreting two particular requirements. These were 1.2.1 FCS Classifications and 1.2.3 FCS Criticality. These requirements were reviewed and recommended for revision several times over the span of the validation. Each reviewer contributed something worthwhile to clarify the requirements, but there was usually disagreement about their meaning. Finally, changes were recommended for these requirements which are logical, comprehensible and compatible with the FCS design development process and the intent of Specification MIL-F-9490D.

PREFACE

This report was prepared by the Lockheed-Georgia Company, Marietta, Georgia, for the Air Force Flight Dynamics Laboratory under USAF Contract F33615-76-C-3034, Project No. 1987. Thomas D. Lewis was the Project Engineer/Technical Monitor.

Mr. Ralph J. Hylton served as the principal investigator for this program. Other Lockheed personnel who made major contributions were Messrs. Charles W. Kettering, John M. McCarty, Harold A. Valery and William E. Jordan.

The validation results are reported in three volumes as follows:

Volume I - Summary of YF-17 and C-5A Validations

Volume II - YF-17 Lightweight Fighter Validation

Volume III - C-5A Heavy Logistics Transport Validation

The contractor's report number is LG77ER0010. This report covers work from April 1976 to January 1977. It represents the view of the authors, which are not necessarily the same in all cases as the views of the Air Force. This report was submitted by the authors January 19, 1977.

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SECTION I

INTRODUCTION

This report is prepared as part of a continuous effort by the Air Force Flight Dynamics Laboratory, Wright Patterson Air Force Base, Ohio, to update and improve Military Specification MIL-F-9490, "Flight Control Systems - Design, Installation and Test of Piloted Aircraft, General Specification For." The specification contains requirements that are applied by the aircraft industry in design, development, and ground and flight test demonstrations of new airplanes.

Section II of this volume presents brief descriptions of the Lockheed C-5A heavy logistic transport airplane and its flight control system (FCS). The C-5A is an operational long-range, all-weather, high-altitude, high-subsonic heavy logistics transport. Its flight control system was designed to meet the system requirements set forth in CP 40002-6B, "Performance/Design and Product Confirmation Requirements for C-5A Air Vehicle, Flight Control Subsystem," which were necessary to accomplish the missions defined for the C-5A. The basis for this specification was MIL-F-9490C and would be equivalent to the controls specification required by MIL-F-9490D, Paragraph 4.4.2.

The validation methodology, presentation format and individual validations are presented in Section III of this volume. Each C-5A applicable requirement and other requirements of particular interest to Lockheed have been subjected to the validation process. The evaluations were based on C-5A requirements and existing C-5A ground test, flight test and analytical data. The depth of validations varied dependent upon the availability of data. C-5A data, however, were generally plentiful and sufficient for the thorough evaluations conducted.

From the validation process, conclusions about the applicability of each requirement to the C-5A and the next generation transport aircraft FCS with respect to practicability, accuracy and completeness were developed. To insure to the reader the necessary insight into the validation process and to justify conclusions and recommendations about the C-5A and its FCS characteristics and operation in many individual requirements validations, a tabular summary of the C-5A validation study is presented in the Summary. It identifies each particular requirement validated as to whether a change was recommended, what level of C-5A compliance was established, whether the stringency was considered suitable and whether or not we recommend additional data to be included in the "Users' Guide."

Recommendations to improve the specification and "Users' Guide" are presented in Section V. The recommendations are based on the realization that the specification strives for cost effective FCS which permit the aircraft to satisfy the USAF mission performance reliability requirements and provide the desired flying qualities during normal operations and after failures.

SECTION II

AIRPLANE DESCRIPTION

1. General Physical Characteristics

The C-5A is a Class III (heavy logistic transport) airplane as classified in Paragraph 1.3 of MIL-F-8785B. It is a long range, all weather, high altitude, high subsonic, swept wing, T-tailed airplane with relatively short field performance capability. The C-5A is powered by four General Electric TF-39 turbofan engines equipped with thrust reversers. Inflight reverse thrust is available from the inboard engines for rapid or emergency descent. The C-5A basic configuration and dimensions are shown in the general arrangement, Figure II-1.

The aircraft gross weight ranges from 319,809 lbs. empty to 769,000 lbs. maximum design gross and can carry up to 265,000 lbs. payload of a wide variety such as heavy-wheeled combat support equipment and personnel. Aerial delivery of single package payloads of 86,000 lbs. has been demonstrated and up to 200,000 lbs. may be air dropped in multiple packages when the aft cargo door is opened.

A retractable, high flotation landing gear system is provided and contains steerable nose gear and main gear which can be set "crabbed" for crosswind take-off and landing.

Hydraulic and electrical power are supplied from each engine for normal flying. Two auxiliary power units supply electrical, pneumatic and hydraulic power for engine starting and for ground operation, maintenance and systems check-out.

2. Flight Controls

2.1 General

The C-5A flight control functions are listed and generally defined below.

<u>Manual Flight Control Systems (MFCS)</u>. MFCS are those using pilot commands as the primary action to initiate control system activity to provide changes in airspeed, control forces and moments necessary to produce changes in altitude, heading, attitude and flight path. MFCS functions include pitch, roll, yaw, side force, lift, drag, trim and thrust.

<u>Aerodynamic Enhancement Flight Control Systems (AEFCS)</u>. AEFCS are those systems which improve ride qualities, improve stability of the aircraft or augment the pilot's ability to control.

<u>Automatic Flight Control System (AFCS)</u>. AFCS are those systems providing automatic maintenance of or diversion from established flight path condition and/or providing dedicated displays for pilot primary control of the flight path or for monitoring automatic control. AFCS provides automatic activity primarily

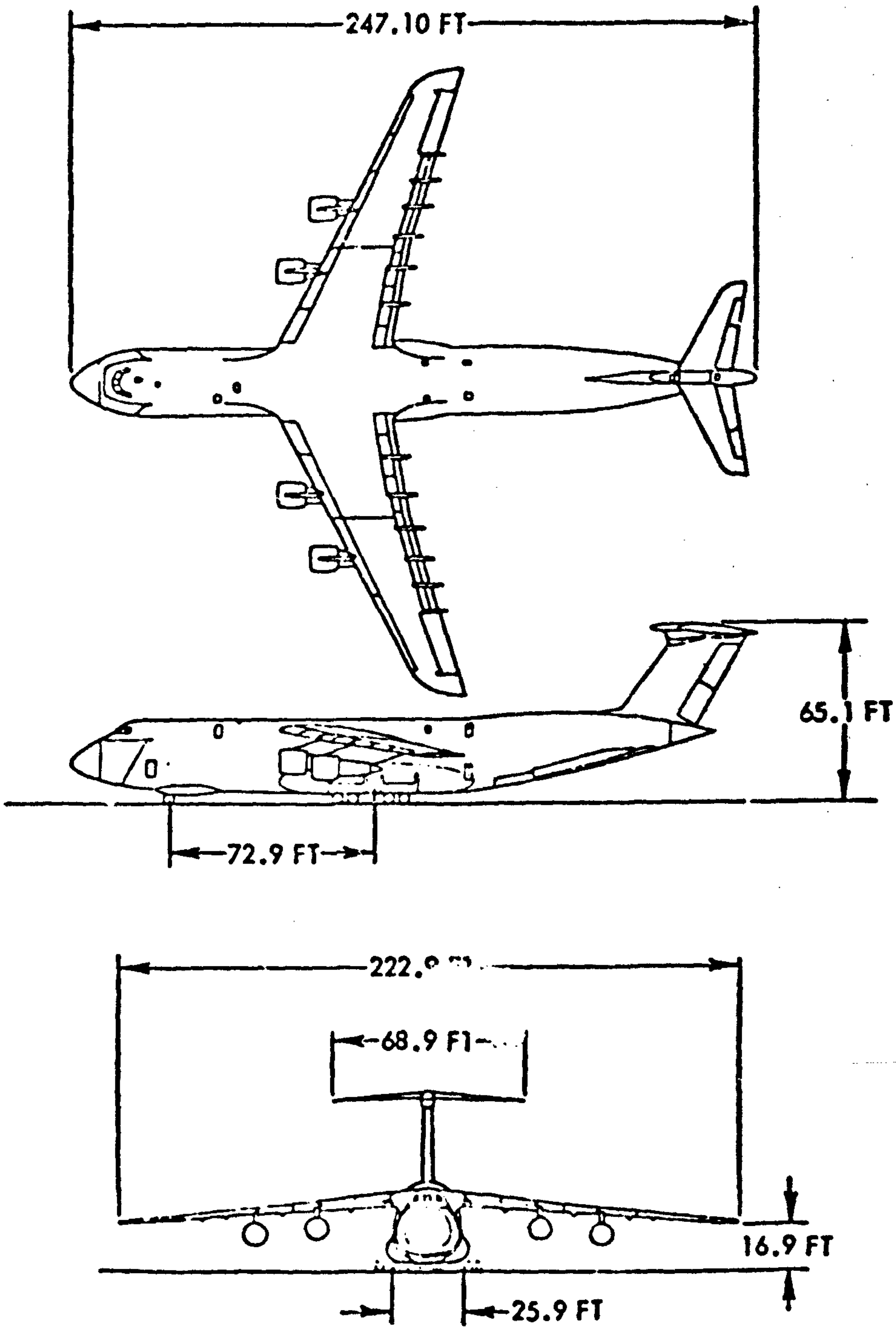


FIGURE II-1. C-5A GENERAL ARRANGEMENT

independent of pilot commands except as required for control wheel steering or to activate, deactivate, pre-select or reselect modes of operation. AFCS provide automatic control of such preselected flight conditions as airspeed, altitude, attitude and heading. AFCS may also provide automatic flight path control such as terrain following and precision course direction (auto land and auto nav). The AFCS includes autopilots, autothrottles, flight directors (including flight instruments), and similar control subsystems.

Limiting Flight Control Systems (LFCS). LFCS are those FCS which provide structural load alleviation or flutter suppression. These controls may act automatically to reduce the combined loads effects of maneuvering when encountering external disturbance (gusts and turbulence). Also they may provide fixed or varying degrees of aerodynamic damping necessary to assure overall flutter-free operation.

Table 1 shows the flight control surfaces, devices and subsystems which provide these control functions. The control functions are powered from four independent hydraulic systems and four electrical power systems which provide AC or DC power appropriate for each application. The hydraulic power distribution is shown in Figure II-2 and the electrical power distribution is shown in Figure II-3. The FCS functions and operation are discussed in greater detail in the following paragraphs.

2.2 Roll Axis Flight Control System

Roll axis control subsystems are listed below irrespective of their classifications:

- Ailerons
- Spoilers
- Manual Trim
- Mechanical Feel
- Stability Augmentation (SAS)
- Active Lift Distribution (ALDCS)
- Autopilot

Roll control is provided by ten flight spoilers operating differentially in conjunction with two conventional ailerons. These spoilers also deploy symmetrically as ground spoilers after touchdown.

The spoiler/aileron interface is shown on Figure II-4. A mix box in each wing converts an input signal from the aileron cable system to a shaped input to the flight spoilers. The same mix box relays an input signal from the ground spoiler cable system to the flight spoilers when they function as ground spoilers.

The aileron servos respond to mechanical inputs from the pilots, autopilot, and a series of electromechanical trim actuator located in the linkage to each

TABLE 1
C-5A FLIGHT CONTROL (FC) FUNCTIONS

Flight Control Function	Surfaces/Devices and Subsystems
MFCS:	
Maneuver	All Elev., Var. Feel, and Elevator Manual Controls Ailerons, Spoilers, and Aileron Manual Controls Both Rudders and Rudder Manual Controls
Trim	Horizontal Stab. (Normal & Emer.) Ailerons and Trim Controls Both Rudders and Trim Controls (Norm) Both Rudders, Yaw Aug. Man. Trim (Emer.)
Lift/Drag	T.E. Flaps, L.E. Slats, Ground Spoilers
Thrust	Throttles Control System
AEFCS:	
Stability Aug.	Inboard Elevators and SAS Ailerons and SAS Both Rudders and SAS
Stall Warning	Stick Shaker and Audible Warning
Stall Limiting	(None)
AFCS:	
Auto Control	All Elevators, Horiz. Stab. and A/P (Auto Pilot) Systems Ailerons, Spoilers and A/P System Both Rudders and A/P System
Automatic Thrust	Auto-Throttle System
Dedicated Displays	Flight Director and Flight Instruments Angle of Attack System and Instruments
LFCS:	
Load Control	Inboard Elevators, SAS and ALDCS Ailerons, SAS and ALDCS
Limiting	Both Rudders and Rudder Limiter

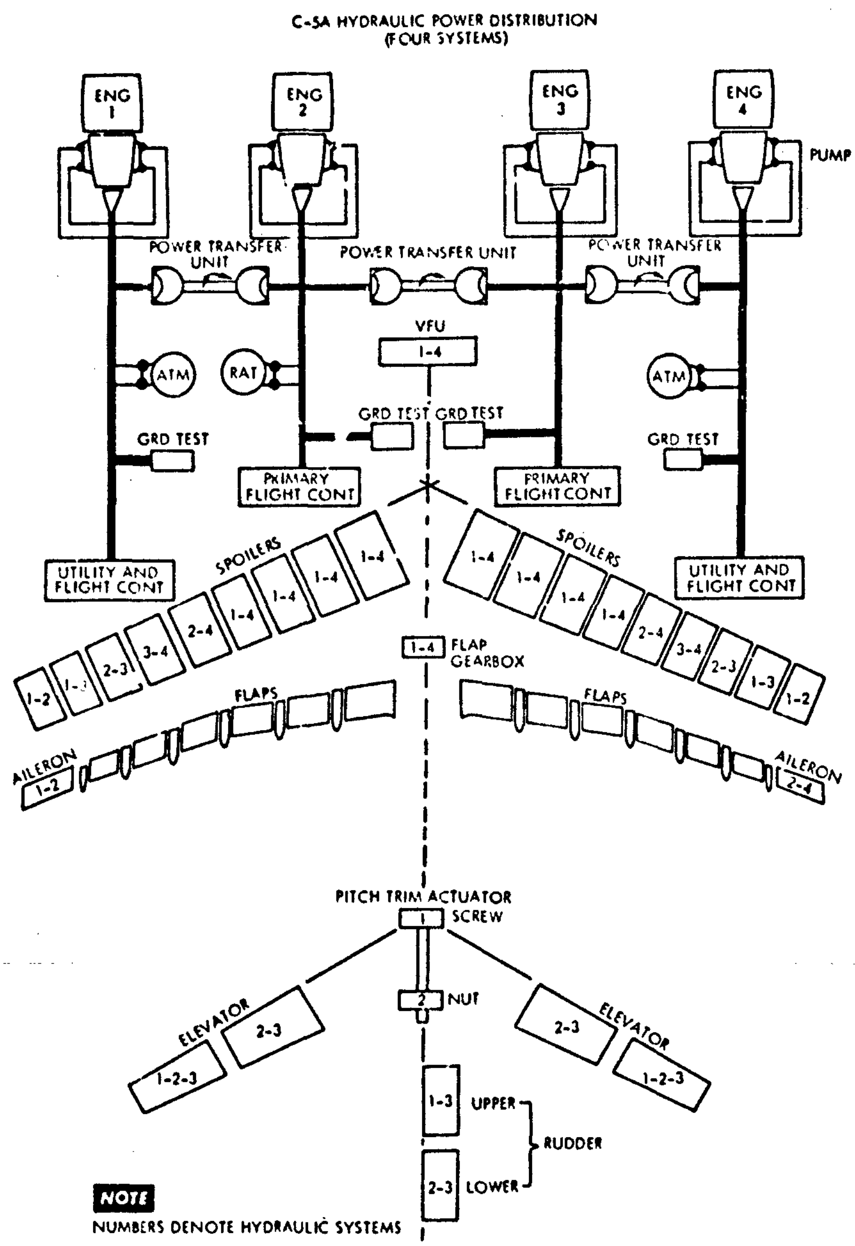


FIGURE II-2. C-5A FLIGHT CONTROL HYDRAULIC POWER DISTRIBUTION

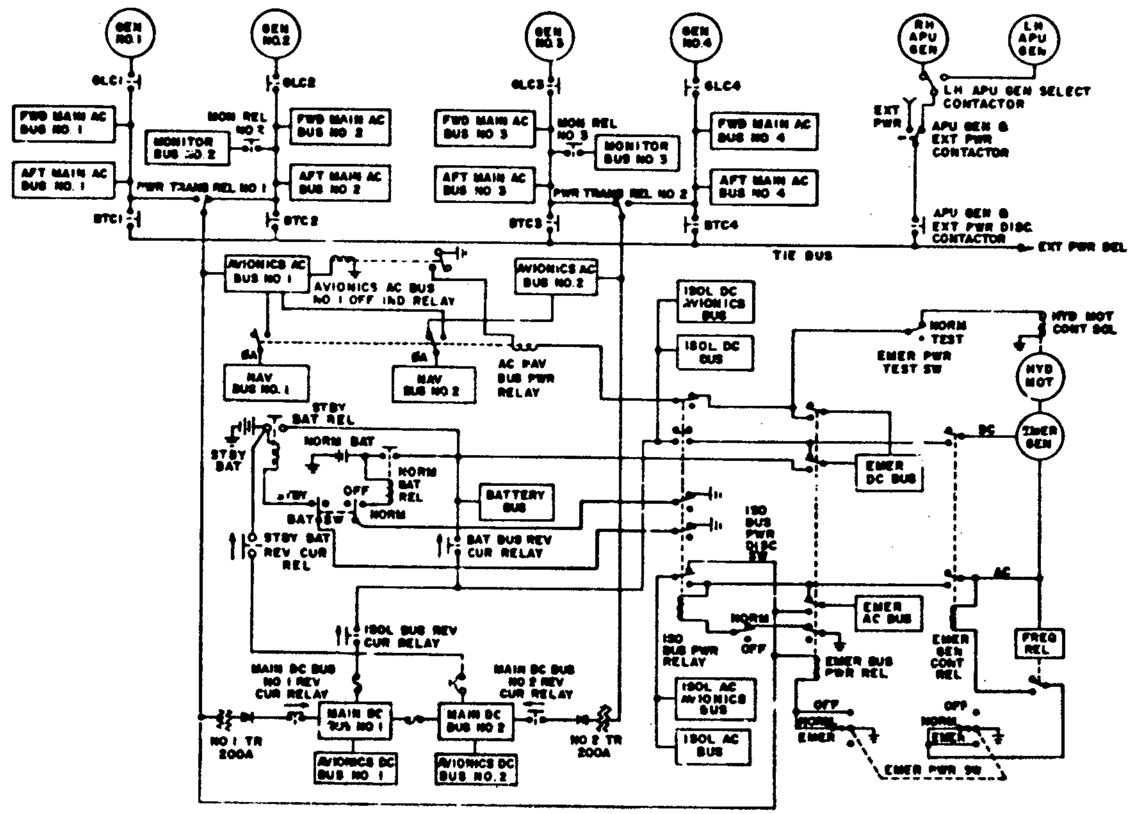


FIGURE II-3. C-5A ELECTRICAL SYSTEM

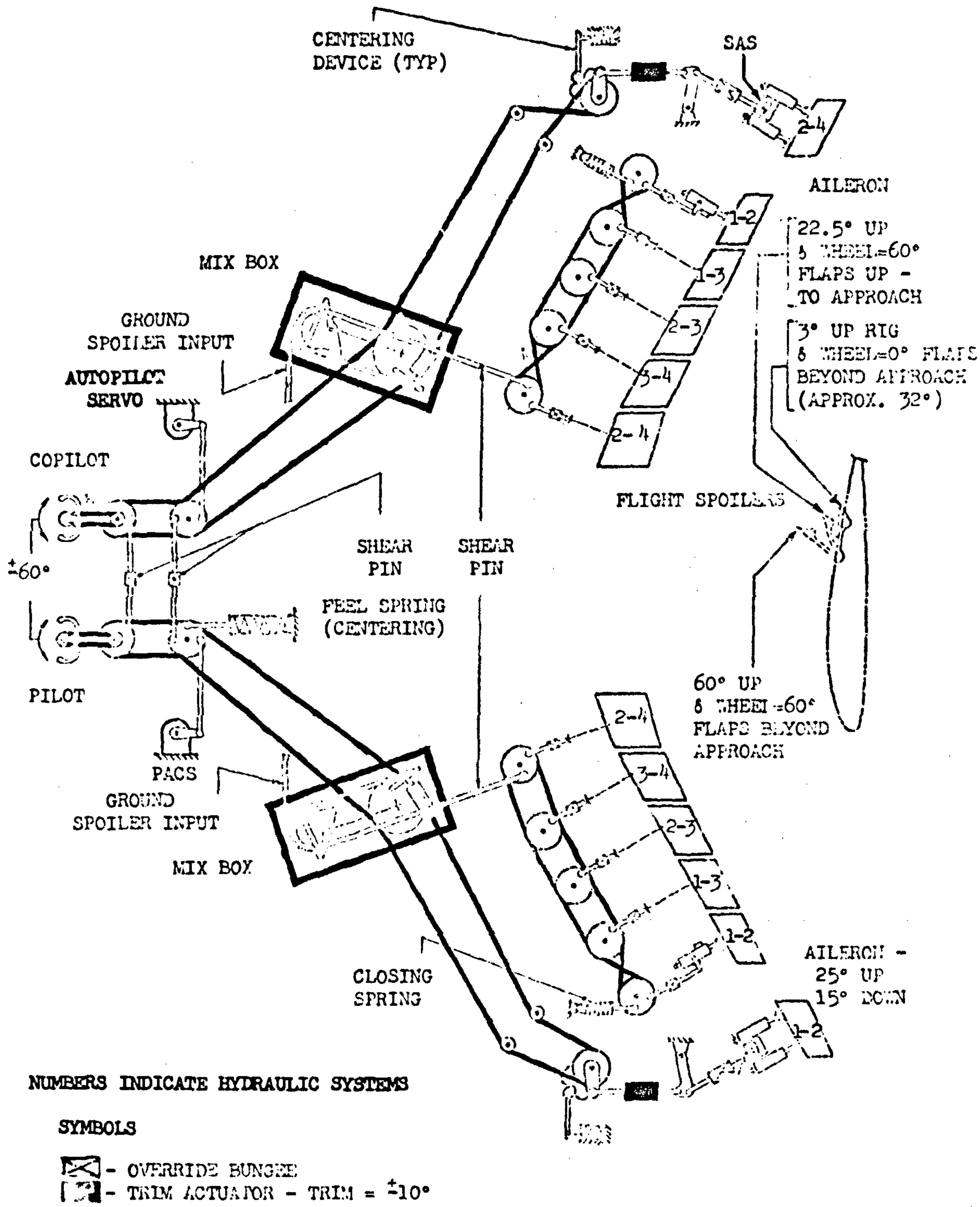


FIGURE II-4. C-5A ROLL CONTROL SYSTEM

servo. Electrical inputs from an automatic flight control computer acting through a series dual hydraulic Stability Augmentation System (SAS) servo within each aileron servo provide for lateral stability augmentation and active lift distribution control.

The spoiler servos respond to mechanical inputs from the pilots, autopilot, and an electromechanical up-rig/shifter actuator located in each mixer. When the trailing edge flaps extend to approximately 32 degrees, an electrical signal to the ratio shift actuator up-rigs the flight spoilers 3 degrees from the faired position. This is done to minimize the loss of spoiler-flap lift with roll control.

A simple mechanical feel/centering spring is attached to the roll control input system. Additional feel is obtained from the flight spoiler closing springs attached to the outboard flight spoiler input quadrant and a centering cam assembly located on the cable quadrant at each aileron servo.

An electromechanical trim actuator is located in the input linkage of each aileron control servo assembly and is in series with the pilot input system. Operation of the aileron trim knob, located on the center console, results in retraction or extension of the trim actuator, which serves as a mechanical input to the aileron control servo assembly. Each aileron trim actuator may be energized separately, by operating a switch located to the side of the aileron trim knob, to provide roll trim in the event one actuator is inoperable. An indicator with dual pointers, located in the flight station, indicates the position of each aileron relative to the faired position.

The roll control system includes conventional pilot's and copilot's control wheels with rotation from neutral to ±60 degrees. Corresponding control surface deflections are shown in Figure II-4. The SAS, ALDCS and autopilot subsystems will be discussed for all axes later in this section.

2.3 Pitch Axis Flight Control System

Pitch axis control subsystems are listed below irrespective of their classifications:

- o Inboard Elevators
- o Outboard Elevators
- o Manual Trim
- o Variable Feel
- o Stability Augmentation (SAS)
- o Active Lift Distribution Control (ALDCS)
- o Stallimiter
- o Autopilot

Pitch control is provided by means of four separate elevator surfaces hinged at the rear beam of the horizontal stabilizer. The elevator control system is shown schematically in Figure II-5. Control column motion is transmitted through a cable system to the full-power, irreversible type hydraulic servos

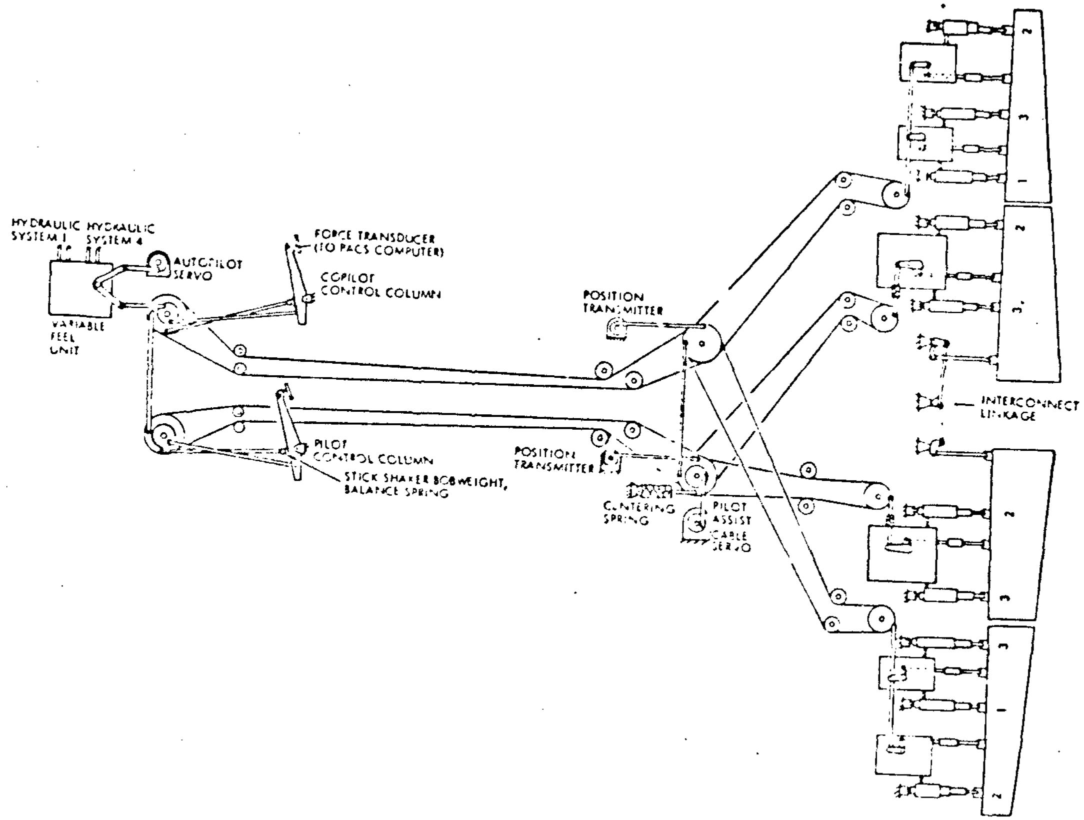


FIGURE II-5. ELEVATOR CONTROL SYSTEM

which power each surface. Inboard elevator surfaces are structurally interconnected by a mechanical linkage, and each is powered by a dual actuator servo package. Each outboard surface is powered by a triple hydraulic servo package. Normally, the left hand inboard elevator is powered by system 2 and the right hand inboard elevator is powered by system 3. The pilot can switch on the inactive system after a hydraulic system failure to maintain power to both surfaces. The outboard elevators are normally powered by hydraulic systems 1, 2, and 3.

All four elevator servos respond to mechanical inputs from the pilots and autopilot. The inboard servos additionally respond to electrical inputs from a SAS flight control computer. Each inboard servo incorporates a dual hydraulic series SAS servo.

An elevator artificial feel subsystem provides the pilots and autopilot with appropriate feel forces to permit safe maneuvering of the aircraft throughout its operational flight envelope. This subsystem consists of three types of force-producing components:

1. The system-centering spring plus four servo centering springs

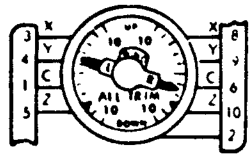
2. The bobweight effects of the control columns and the stick shaker mounted on each

3. The system variable feel unit (varies feel with dynamic pressure)

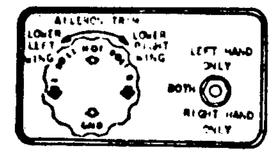
The pitch trim system includes the horizontal stabilizer actuator and its input systems. A high degree of safety is inherent in the system since two electrical input signals are normally required before the actuator can operate and the screwjack is a dual structural path irreversible linear device.

Pitch trim is accomplished by movement of the entire horizontal stabilizer, but its control is independent of the primary pitch control system (elevators). The trim actuating and indicating systems are shown in Figure II-6. Main features of the pitch trim system are listed below:

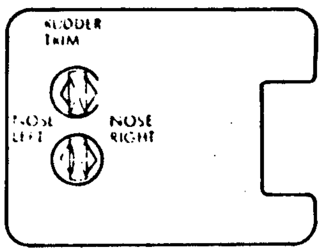
- One pitch trim actuator
- Two electrical command systems
- One mechanical command system
- One autopilot command system to signal the screw drive
- Four horizontal stabilizer position-limit switches
- Two independent hydraulic-powered trim drive gear boxes
- One horizontal stabilizer position indicator system



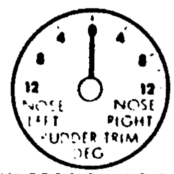
AILERON TRIM INDICATION



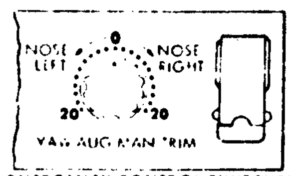
AILERON TRIM CONTROL
CENTER CONSOLE



RUDDER TRIM CONTROL SWITCHES
CENTER CONSOLE (COPILOT S SIDE)



TRIM POSITION INDICATOR



EMERGENCY CONTROL SWITCHES

C-5A PITCH TRIM SYSTEM

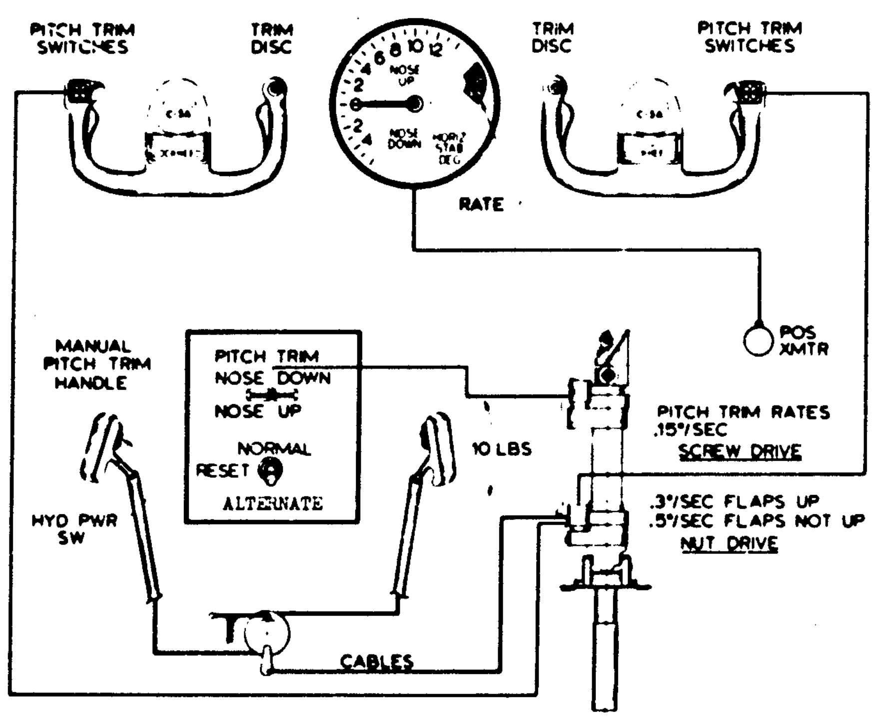


FIGURE II-6. C-5A TRIM CONTROLS AND INDICATION

2.4 Yaw Axis Flight Control System

Yaw axis control subsystems are listed below irrespective of their classifications:

- Upper Rudder
- Lower Rudder
- Manual Trim
- Mechanical Feel
- Rudder Limiter
- Stability Augmentation (SAS)
- Yaw Aug. Manual Trim
- Autopilot

Directional control is provided by upper and lower rudders. Each surface is powered by a dual hydraulic, fully powered servo. The rudder control system is shown schematically in Figure II-7. Hydraulic systems 2 and 3 power the lower rudder and systems 1 and 3 power the upper rudder. Manual maneuvering in the yaw axis is accomplished by displacement of conventional rudder pedals. Superimposed upon the manual input system, in a series fashion, is the SAS, which has the authority of 20.5 degrees of surface travel.

Rudder pedal nose wheel steering allows either pilot to command nose wheel deflection with pedal travel. This control provides assistance to the rudder in yaw axis control of the aircraft during landing and take-off.

A simple mechanical feel/centering spring is attached to the lower rudder input quadrant.

The rudder trim actuator repositions the neutral point of the preloaded feel spring after the rudder pedals have been displaced to a desired trim position. Trim actuator operation is controlled by simultaneous operation of two rudder trim control switches located on the copilot's side of the center console.

Emergency rudder control provides the pilot with $\pm$20 degrees of upper and lower rudder authority. A YAW AUG MAN TRIM control knob is provided on the flight augmentation panel to permit control of the rudders through the yaw augmentation subsystem in the event of a jam in the single rudder cable system. A guarded switch to the right of the control knob must be moved from the OFF position to the ON position before the emergency mode becomes operational.

The rudder position and travel are pedal-limited by mechanical stops positioned by an electromechanical linear actuator, as shown on Figure II-7. The rudder position limiter assembly is installed at the lower rudder input quadrant. The input actuator responds to dynamic pressure and Mach number signals from the air data subsystem.

2.5 Lift and Drag Control System

Lift and drag control subsystems are listed below irrespective of their classifications:

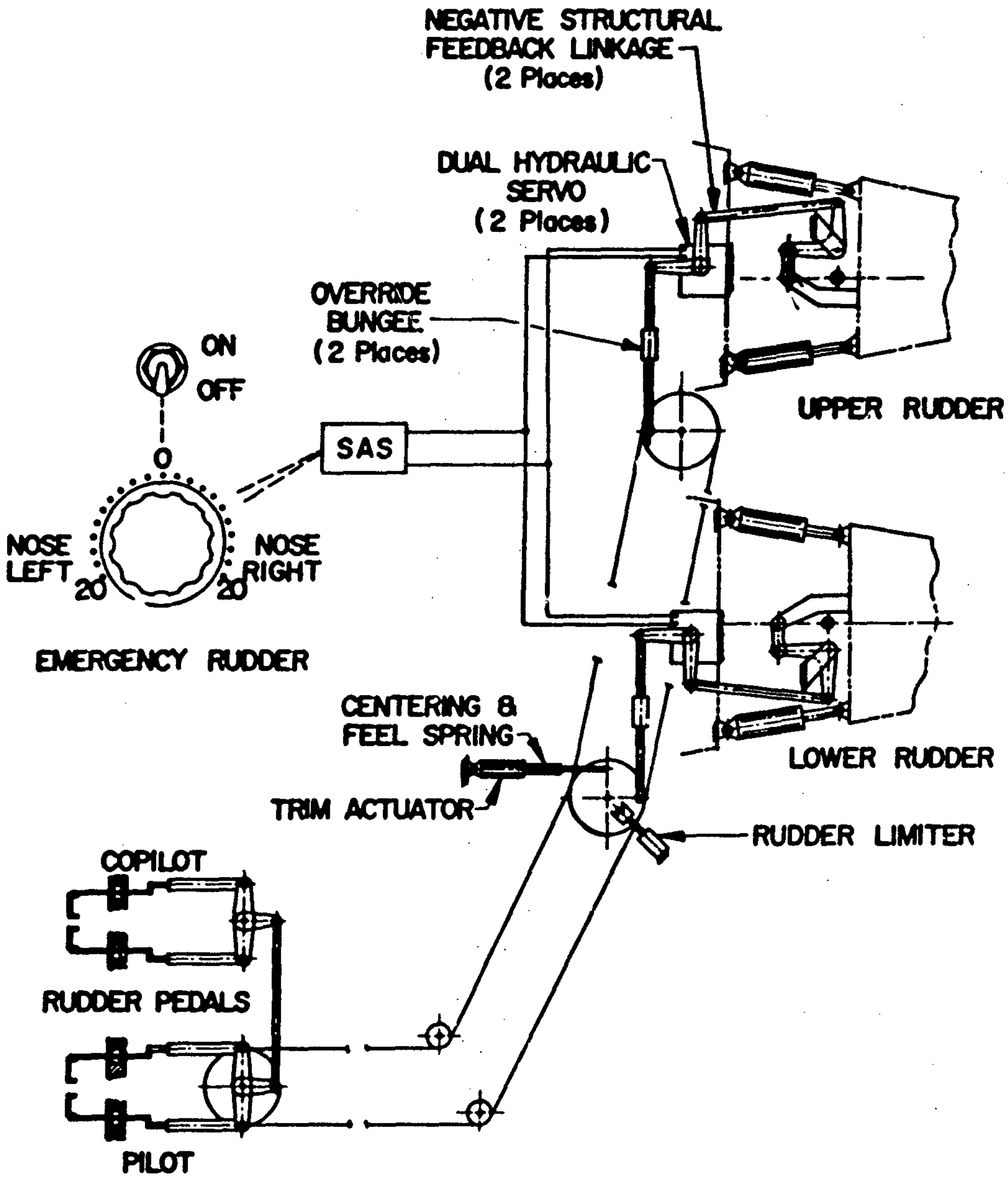


FIGURE II-7. RUDDER CONTROL SYSTEM

- o Trailing Edge Flaps/Leading Edge Slats
- o Ground Spoilers
- o Thrust Reversers

The C-5A employs leading edge slats and Fowler type trailing edge flaps to change the relatively low-lift wing required for high-speed flight to a high-lift wing necessary for short landings and take-offs. Actuation of the flap/slat systems is accomplished by displacement of a single flap control handle located on the center console. Asymmetry detection with test circuitry and position indicators is provided for each system. The panels are positioned by ball screw actuators which drive the flap carriage in each straight track. The actuators are driven through a torque tube system by a dual hydraulic power package. The ball screw actuators are driven through a torque tube drive system by the flaps power package. Major elements of the flap system are shown in Figure II-8.

A ground spoiler system is provided to spoil wing lift and to increase drag, thereby reducing landing and rejected take-off distances. The ground spoilers consist of four inboard panels and five outboard panels per wing (the outboard panels are also used asymmetrically to augment roll control). All ground spoiler panels are deployed symmetrically upon command by either pilot from heavily detented interconnected control handles located on the center console. The detent is removed automatically by either a landing gear touchdown signal or wheel spin-up. The ground spoiler system is shown schematically in Figure II-9.

All ground spoiler panels are commanded only to extend or retract. Each inboard panel is actuated by a simple dual actuator arrangement, each containing a mechanical locking device which is released hydraulically upon command to extend those panels. Each outboard spoiler panel is actuated by a dual servo actuator which permits both symmetric and asymmetric control to any position within the limits.

Thrust reversers and their operation are discussed for convenience with Section II. 2.6 Thrust Control Systems.

2.6 Thrust Control Systems

Thrust control subsystems discussed in this section include manual control of engine thrust and reverse thrust. Automatic throttles provided for the C-5A are discussed for convenience in Section II. 2.7 Automatic Controls Systems. Each pilot is provided with a set of four independent throttle control levers. Both sets of four levers are interconnected. Thrust directions and levels are selected by movement of these throttle control levers which operate a separate independent conventional mechanical system extending from the flight station to each engine fuel control lever.

The thrust control system includes stops, adjustments and interlocks to minimize the potential for undesired power applications. These included the following:

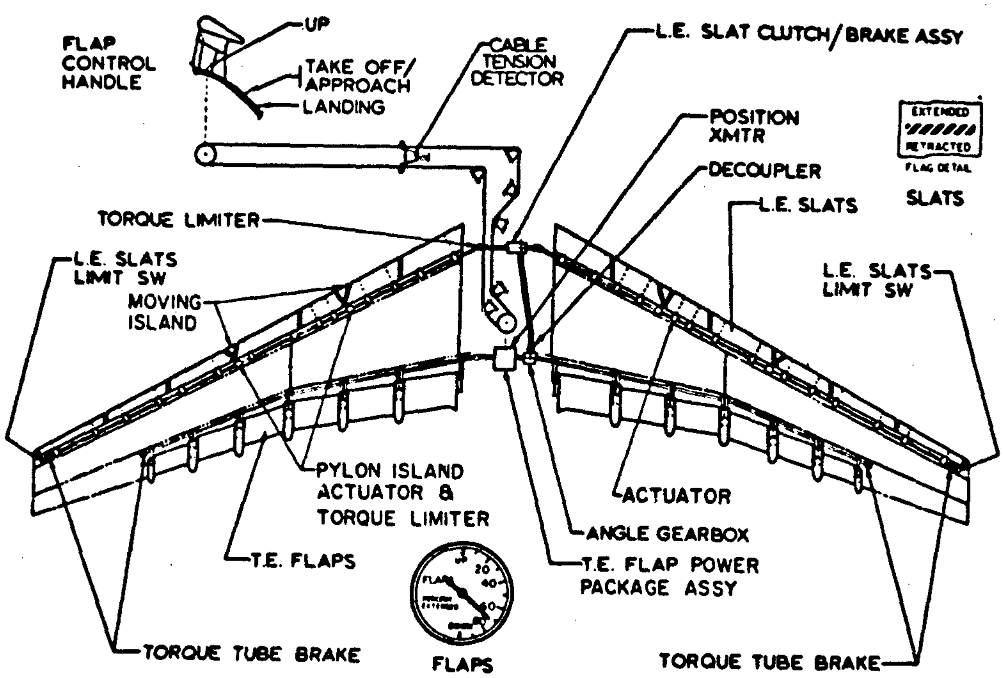


FIGURE II-8. C-5A HIGH LIFT SYSTEM

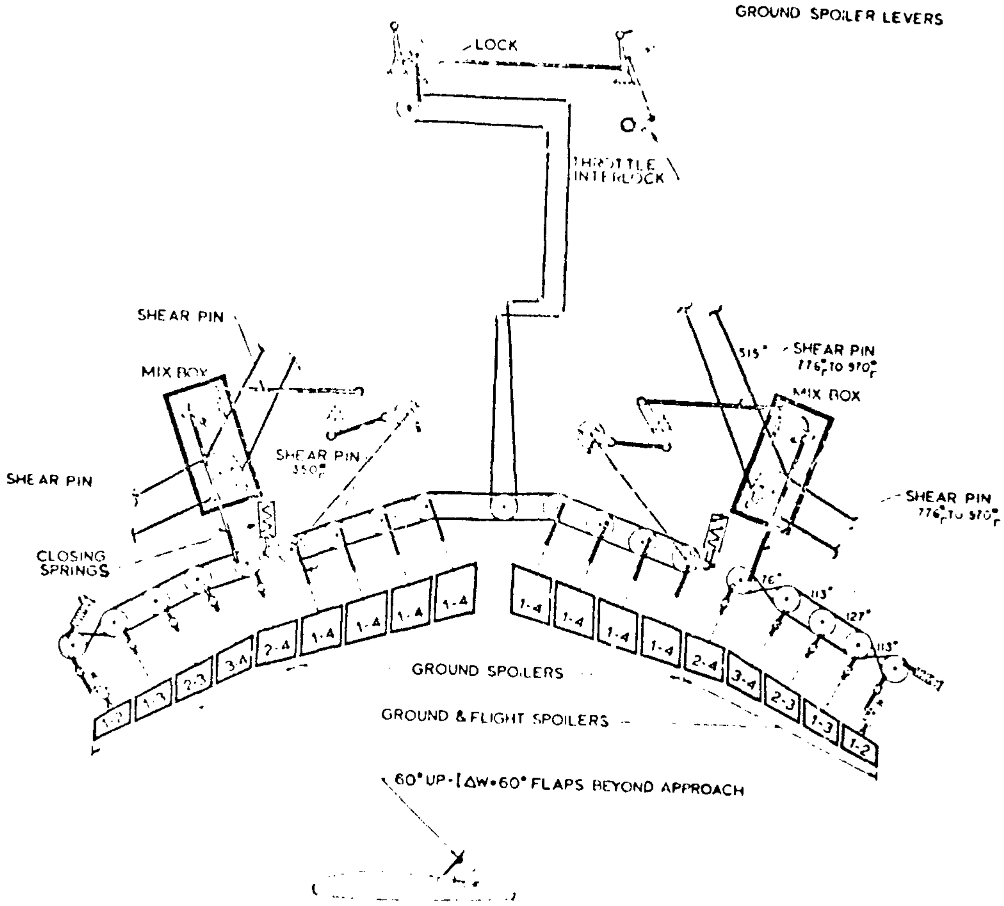


FIGURE II-9. C-5A GROUND SPOILER SYSTEM
(LIFT, DUMP AND DRAG)

- Maximum forward and reverse thrust stops
- Reverse thrust limiting adjustment
- Throttle control lever friction adjustment
- Landing-gear-not-extended warning (min. cruise)

The thrust control system provides for inflight thrust reversing of the inboard engines and thrust reversing of all engines for short field landing and rejected take-off. The C-5A thrust control system is shown schematically in Figure II-10 and its component routing and pilot control placements are shown in Figure II-11.

2.7 Automatic Controls Systems

The subsystems presented in this section include, for convenience, those which either provide warning of impending vehicle stall, augment vehicle stability, improve vehicle ride qualities, dedicated displays, provide automatic flight control functions and provide structural dynamic load alleviation or limiting. In the C-5A, the pilot is retained as the major system manager in the automatic control loops. The pilot can, at any time and during operation of any automatic control mode, take command and manually control the aircraft to complete the mission. C-5A automatic control subsystems are shown in Figure II-12.

The C-5A automatic controls are designed for a high mission reliability, fail-safe operation, and to be (in some cases) fail-operational. These requirements are met through the use of various redundancy and monitoring techniques. Fail-safe operation provides system mode or function disengagement or total system disengagement after a first system failure which degrades system performance below established levels. Another form of fail-safe operation provided results from the pilot's ability to override or overpower a system action at any time. Fail-operational capability provides continued system operation with full authority after a first system failure which would have degraded system performance below established acceptable levels. This is accomplished through redundancy circuit switching. The capabilities of the automatic controls are:

- Automatic stall-warning
- Automatic throttle functions
- Automatic pilot basic functions
- Automatic pitch trim control
- Automatic enroute navigation (VOR, TACAN, and inertial)
- Automatic terrain following (vertical flight path control)
- Automatic terminal navigation (ILS approach, radar approach, and air drop)
- Automatic landing (flare, throttle retard, and rollout)
- Automatic go-around (vertical flight path control)
- Three axis stability augmentation
- Active lift distribution
- Dedicated pilot and copilot displays

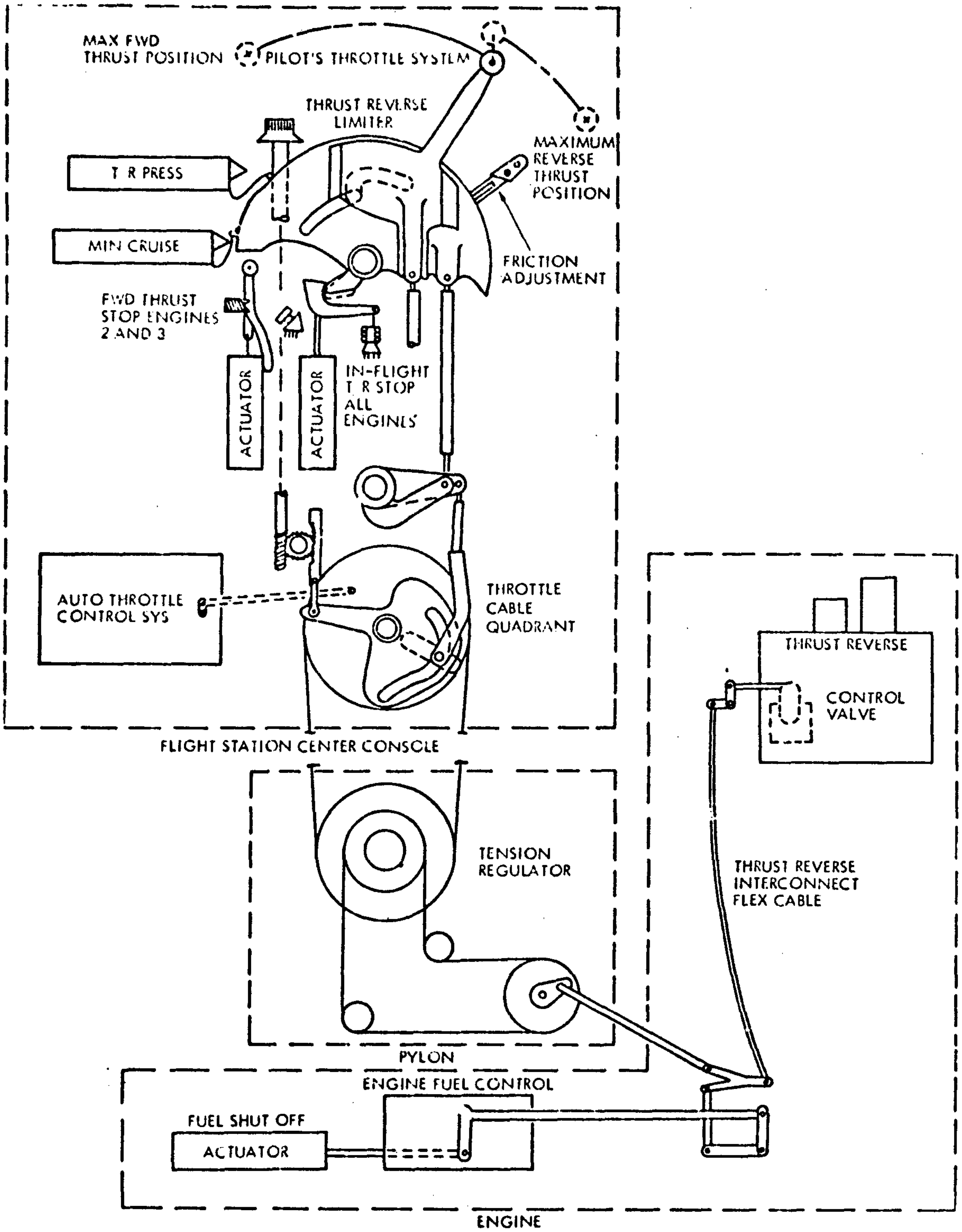
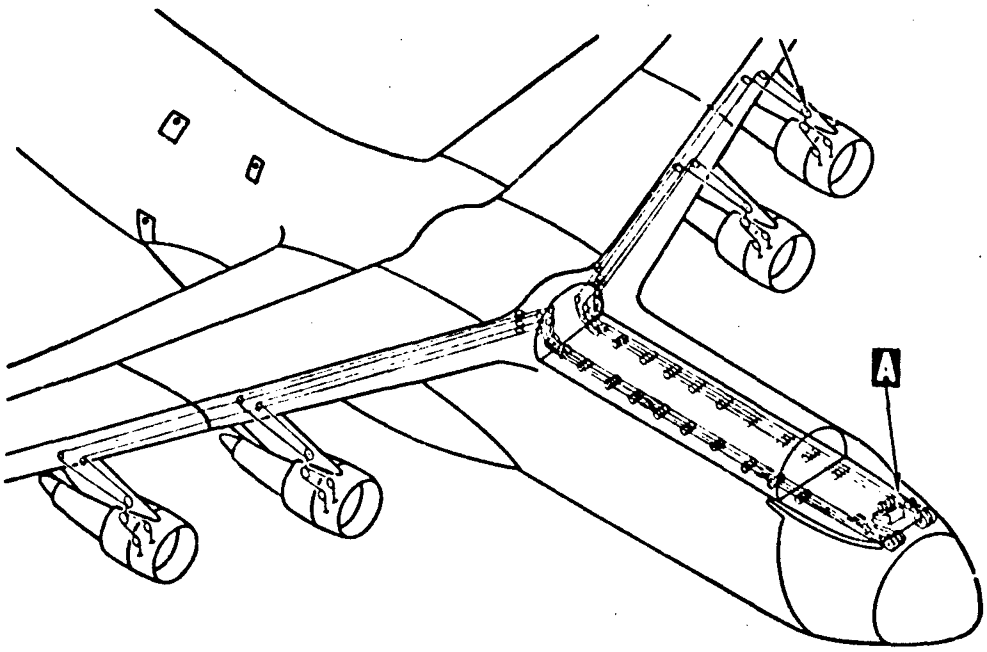
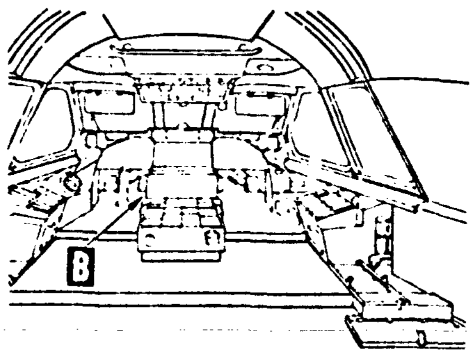


FIGURE II-10. C-5A THRUST CONTROLS

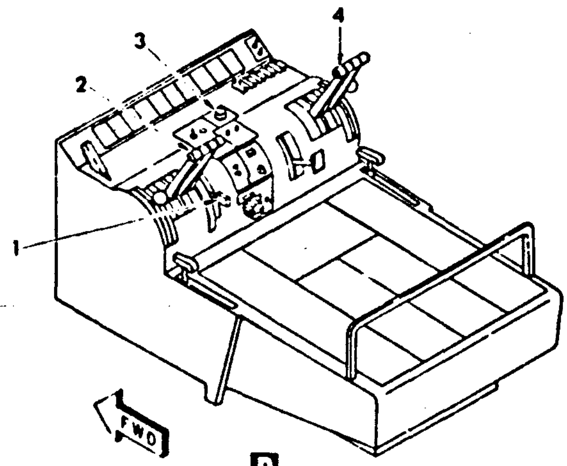




A

PILOTS' STATION

1. THROTTLE FRICTION LEVER
2. PILOT'S THROTTLE LEVERS
3. REVERSE THRUST LIMITER KNOB
4. COPILOT'S THROTTLE LEVERS



B

CENTER CONSOLE

FIGURE II-11. C-5A THRUST CONTROLS, PLACEMENT AND ROUTING

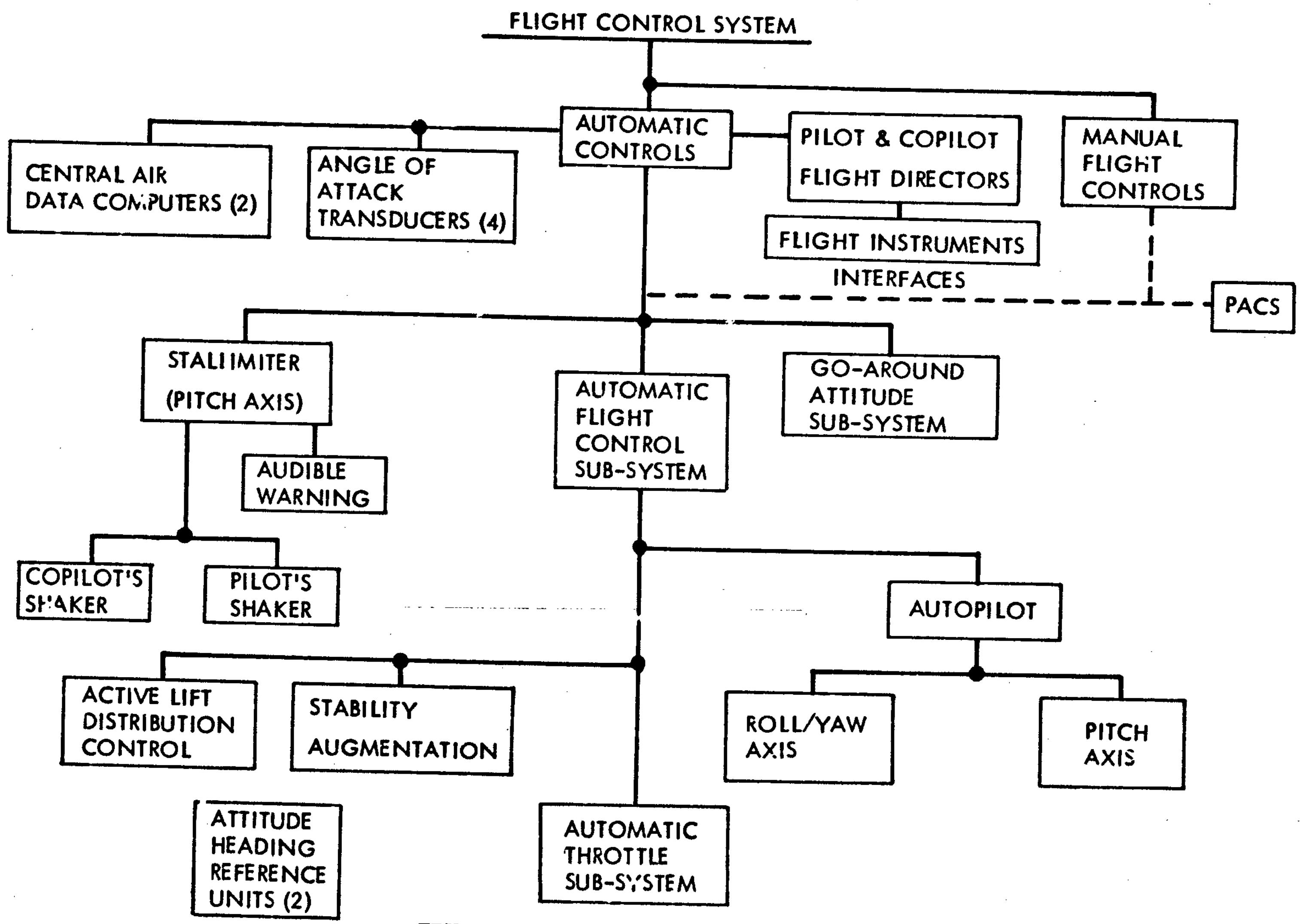


FIGURE II-12. AUTOMATIC CONTROLS SUBSYSTEM

Operation of all automatic control systems is controlled from the flight station by means of various control and test panels located on the center console, throttle quadrant, pilot's and copilot's side consoles, pilot's overhead panel, and navigator's panel.

Two independent attitude heading reference units (AHRU) are installed on the C-5A. The AHRU supplies roll and pitch attitude and heading information to the automatic flight controls and other aircraft navigational subsystems. The AHRU may be used as a flux valve compass, directional gyro compass, or a magnetic slaved directional gyro stabilized compass. Roll and pitch information is available regardless of the azimuth mode of operation.

There are four angle-of-attack transducers (AOAT) on the aircraft, two on each side of the fuselage above and just aft of the nose landing gear. These transducers are actuated by a vane extending into the airstream which measures the angle of the relative wind flowing across the vane. This information is fed to the stallimiter, go-around attitude, and automatic throttle systems for use in their computations. The only other component in the AOAT system is the AOAT anti-ice panel located on the pilot's overhead panel. This panel supplies power to the vane heaters and will indicate a heater fault.

The stallimiter is a stall warning system which provides the pilot and copilot with a warning of an impending stall condition. Two identical channels identified as STALLIMITER 1 (pilot's) and STALLIMITER 2 (copilot's) are provided. One control column shaker per channel provides the stall warning. Each stallimiter channel contains dual redundant comparator circuits for fail-safety.

The primary function of the go-around attitude system (GAAS) is to provide optimum rotation and climb-out pitch steering commands for manual or automatic go-around control. Its secondary function is to furnish fuselage reference line angle-of-attack signals for display to the pilots and for use by the multimode radar during terrain-following. The GAAS consists of one dual channel computer and two go-around mode engage/disengage switches. Full-time operation of the system is required and no controls are provided except the two go-around mode switches on the pilot's and copilot's outer control wheel grips. The system becomes operational when power is applied to the aircraft buses.

The automatic throttle system (ATS) provides th .ttle control for remote throttle positioning, maintaining a desired IAS or Mach number, and maintaining

a desired angle of attack. It also provides throttle retard during the automatic flare maneuver and throttle advance during a go-around maneuver. The ATS consists of one dual-channel computer, two servo motor assemblies (one for the inboard engines and one for the outboard engines), and various controls and switches. The AFCS control panel is shared with the ATS to provide control and test functions. Disengage switches are provided on each pilot's outer throttle lever. Throttle friction must be set to minimum for ATS operation.

During ATS operation, each throttle can be manually adjusted by the pilots for engine power trimming. Also, the pilots can manually override all four throttles simultaneously with the ATS engaged if the need should arise. Maximum throttle travel limiting is provided to prevent engine overboost. Minimum throttle travel limiting is provided to prevent excessive power reduction. Idle disconnect is provided to disengage the ATS if two or more throttles are moved to the idle position.

Stability augmentation is provided for all three axes of the C-5A. The yaw augmentation (Y/A) and lateral augmentation (L/A) systems provide more than the basic rate damping that is normally provided by augmentation or damper systems, while the pitch augmentation (P/A) system provides only rate damping. The Y/A provides stabilization in the directional axis (yaw damping and dutch roll damping) and turn coordination. In addition, the Y/A also provides manual trim of the rudders in the event of a control cable jam or break. This capability is provided as a backup means of rudder control. The L/A provides roll damping, dutch roll damping, and spiral divergence control. The P/A improves the airplane short-period frequency without appreciably decaying the response to a pilot command.

All the augmentation systems are fail-operational and fail-safe. A first failure will illuminate FAULT lamps on the annunciator panel and the master CAUTION light. The second failure will illuminate an INOP indicator on the annunciator panel and the master CAUTION light which informs the pilot that the system is disengaged.

The system is composed of the two augmentation computers (LRU's), the flight augmentation-control panel, various input sensors, and several components located on the individual hydraulic power units. All three systems are intended for full-time use in either manual or automatic flight. An interlock function is provided to the autopilot which will cause autopilot disengagement should the augmentation disengage, since certain inner-loop autopilot functions are accomplished by the SAS.

The autopilot system (A/P) provides automatic control of the aircraft flight path for basic flight, enroute navigation, low-altitude terrain following, ILS approach and landing (including flare and rollout), go-around, and terminal navigation using radar (includes radar approach and air drop). Functions provided by the A/P are:

1. Basic Control Functions

   a. Attitude hold (pitch and roll)
   b. Heading hold
   c. Normal maneuvering (pitch, roll, and heading)
   d. Control wheel steering (pitch and roll)
   e. Altitude hold
   f. Altitude capture
   g. IAS hold and adjust on pitch
   h. Mach hold and adjust on pitch

2. Enroute Navigation Functions

   a. Heading select
   b. Radio navigation (VOR and TACAN)
   c. Inertial doppler navigation
      Inertial heading
      Destination steering
      Course line
      Vertical navigation (point or slope)
   d. Terrain following on pitch

3. Terminal Navigation Functions

   a. ILS approach and landing, including flare and roll-out
   b. Radar approach and air drop
   c. Go-around

The A/P consists of a Pitch/PACS computer, Roll/Yaw/PACS computer, elevator servo, aileron servo, two control wheel hub assemblies, AFCS control panel, and two A/P disconnect switches located on the outer grip of both control wheels. The pilots can manually overpower or countermand the control action of any A/P axis or control function. This is a fail-safe feature.

Two Flight Director Systems (FDS) are installed on the C-5 airplane, system No. 1 for the pilot and system No. 2 for the copilot. Each system is comprised of the following LRU's:

- Flight Director Computer (FDC)
- Attitude Director Indicator (ADI)
- Horizontal Situation Indicator (HSI)
- Remote Horizontal Situation Indicator Control Panel (RHSI)
- Peripheral Command Indicator (PCI)
- Navigation Selector Panel (NSP)
- Auxiliary Navigation Select Panel (ANSP)

The FDS provides the integrated display data required for manual instrument flying and for visual monitoring during automatic landing approaches and other autopilot modes. Fifteen different modes of operation are available for use by the FDS, including the FD Self Test mode. These modes are defined as follows:

- Manual Heading (HDG)
- Inertial Heading/Destination Steering (IH/DS)
- Visual Omni Range (VOR): Cruise (CRS) and Approach (APP) Configurations
- Tactical Air Navigation System (TACAN): Cruise (CRS) and Approach (APP) Configurations
- Station Passage (SP) (Associated with the VOR and TACAN modes)
- Instrument Landing System (ILS): (Available from two independent sources, ILS-1 or ILS-2)
- Course Line (CL), also known as Track Steer (Available from two independent sources - Primary Guidance Computer or Auxiliary Guidance Computer)
- Terminal Navigation (TN) also known as Air Drop (Available from two independent sources - Primary Guidance Computer or Auxiliary Guidance Computer)
- Vertical Navigation (VN) (Available from the Primary Computer only)
- Airborne Radar Approach (ARA)
- Terrain Following
- Go-Around (GA)
- Altitude Hold (AH) (Available from CADC No. 2 only)
- Navigation Aids Off (Nav. Aids Off)
- Flight Director Self Test (FD ST)

The pilot or copilot can independently select the mode he desires by operating the respective mode select switch(es) on his NSP and when applicable the ANSP.

The incoming signals from the respective interfacing systems are processed by the Flight Director Computer (FDC) to provide computed pitch and roll command outputs to the ADI and PCI. The FDC incorporates the necessary switching logic facilities for routing the basic deviation signals from the respective input source to the vertical and horizontal presentations on the ADI (ILS symbol and vertical deviation pointer) and the course deviation bar on the HSI.

The ALDCS is configured as a means of reducing fatigue damage on the C-5A wing due to maneuver, gust, and peak-to-peak ground-air-ground load sources. The ALDCS computer supplies commands to the pitch and lateral augmentation computers to provide the symmetric aileron and inboard elevator inputs as a function of aircraft response parameters. The ALDCS consists of a dual-channel computer and four wing-mounted normal accelerometers. The system interfaces the air data computers, pitch autopilot, flight augmentation control panel, MADAR,

flight annunciator panel, touchdown switches, and most importantly the pitch and lateral augmentation computers. The system is designed normally to be engaged and operating throughout the C-5A mission. A fail-safe design is employed. System faults are detected, identified to MADAR, and the system is automatically disengaged with appropriate indication on the pilot's annunciation panel.

SECTION III

VALIDATION OF REQUIREMENTS

Introduction

This section presents the validation of military specification MIL-F-9490D (USAF) by checking the specification requirements utilizing the experience and knowledge derived during the procurement and development of the C-5A Heavy Logistics Transport. Each specification paragraph applicable to the C-5A is presented in sequence, either singly or in logical groups, and validated with regard to practicability, accuracy, and completeness as a requirement for procurement, design, test, and installation of flight control systems for future pilot military aircraft. Specification paragraphs not applicable to the C-5A are therefore not validated, but are listed in this section in their proper numerical position together with the paragraph title and the notation NOT APPLICABLE. For ease of reference the paragraph numbers of the specification are used herein.

Validation Format and Methodology

The validation format is comprised of five specific parts. A description of the possible contents of each part follows:

1. Requirement

   In this part, the paragraph is written exactly as it appears in the specification.

2. Comparison

   In this part, the compliance of the system, subsystem, or component with the requirement is described. Test, analytical, and descriptive data are presented where appropriate.

3. Discussion

   In this part, an opinion of the requirement is given, whether or not there is compliance by the C-5A. If the system, subsystem, or component does not comply, the effect that compliance would have had is discussed. If there are valid reasons why compliance is not necessary or would be undesirable, the reasons are given. Where appropriate, an assessment is made as to whether the requirement is good, too lenient, or too strict.

The requirement is also evaluated to determine whether compliance can be practically demonstrated. If not, a determination is made as to whether it can be modified to make it so. Further, the requirement is evaluated to determine whether the stringency can be justified for future aircraft procurement.

If the requirement is judged valid, but C-5A data do not meet the requirement, the reasons for the discrepancy are provided. If a recommendation to change the requirement is being made, pertinent considerations to support the recommendations are given.

4. Recommendation

If a change is considered necessary to improve the practicability, accuracy, and completeness of the specification, a recommendation is given. The recommendation, if any, is given in this part. If a complete rewrite of the specification paragraph is suggested, it is written in this part in specification language. If only a partial rewrite is recommended, the changes to the existing paragraph only are indicated.

5. Additional Data

If a change is considered necessary to improve or update the "Users' Guide," the text to be inserted into the "Users' Guide" is given in this part.

Requirement

1. Scope. This specification establishes general performance, design, development and quality assurance requirements for the flight control systems of USAF manned piloted aircraft. Flight control systems (FCS) include all components used to transmit flight control commands from the pilot or other sources to appropriate force and moment producers. Flight control commands may result in control of aircraft flight path, attitude, airspeed, aerodynamic configuration, ride, and structural modes. Among components included are the pilot's controls, dedicated displays and logic switching, transducers, system dynamic and air data sensors, signal computation, test devices, transmission devices, actuators, and signal transmission lines dedicated to flight control. Excluded are aerodynamic surfaces, engines, helicopter rotors, fire control devices, crew displays and electronics not dedicated to flight control. The interfaces of flight control systems with related subsystems are defined.

Comparison

The C-5A FCS was designed to meet the system requirements, set forth in CP40002-6B, Performance/Design and Product Confirmation Requirements for C-5A Air Vehicle, Flight Control Subsystem, which were necessary to accomplish the missions defined for the C-5A. The basis for this specification was MIL-F-9490C and would be equivalent to the controls specification required by MIL-F-9490D, Paragraph 4.4.2.

Discussion

This paragraph is adequate.

Recommendation

Retain the requirement as stated.

It is further recommended that requirements relative to air data sensors which are not now included be developed for inclusion in a later revision to the specification.

Requirement

1.2 Classification

1.2.1 Flight Control System (FCS) Classifications

1.2.1.1 Manual Flight Control Systems (MFCS). Manual Flight Control Systems consist of electrical, mechanical and hydraulic components which transmit pilot control commands or generate and convey commands which augment pilot control commands and thereby accomplish flight control functions. This classification includes the longitudinal, lateral-directional, lift, drag and variable geometry control systems. In addition, their associated augmentation, performance limiting and control devices are included.

1.2.1.2 Automatic Flight Control Systems (AFCS). Automatic Flight Control Systems consist of electrical, mechanical and hydraulic components which generate and transmit automatic control commands which provide pilot assistance through automatic or semiautomatic flight path control or which automatically control airframe response to disturbances. This classification includes automatic pilots, stick or wheel steering, autothrottles, structural mode control and similar control mechanizations.

Comparison

The C-5A classifies the Flight Control Systems as Primary Flight Controls, Secondary Flight Controls, Automatic Flight Controls and Limiting Controls. This is a different classification than is contained in MIL-F-9490D. The C-5A classifications would not meet the new classifications, but whether the C-5A meets or does not meet this definition is not relevant.

Discussion

The attempt to do away with the old primary and secondary flight control classifications is good. However, including augmentation, performance limiting and control devices under a general classification of manual flight controls is confusing. These systems have traditionally been considered to be automatic controls and the detail design can differ considerably from the other manual controls. It is felt that these automatic controls should be contained under another classification. In addition, the classification of systems within the MFCS should be by function and/or operation.

The AFCS classification states that "semiautomatic flight path control" devices are included. The background information and "Users' Guide" for MIL-F-9490D states that "Semiautomatic control includes flight director functions when the option of automatic or semiautomatic operation is provided." It is not evident from the "Users' Guide" when the flight director is to be included as part of this specification. Requirement 1.1 (Scope) states that this specification includes dedicated displays, and Requirement 3.1.5.1.2 attempts to give some flight director system requirements. It is recommended that this specification should include the flight director system (including flight instruments) requirements. Paragraph 3.1.5.1.2 should be revised and expanded to include these system requirements.

The AFCS classification contained in MIL-F-9490D includes structural mode mechanizations. These systems should be carried under a different classification since their function is very different from autopilot and autothrottle systems in that they have no direct effect on airspeed, altitude, heading, attitude or flight path.

Recommendation

Revise the requirement as follows:

"1.2.1 Flight Control System (FCS) Classifications. FCS are classified as to their function, the role of the pilot in the initiation of their primary control activity, the maintenance of or diversion from established flight conditions, their ability to improve ride qualities or stability, their role in reducing the magnitude of structural loads and improving structural fatigue life, and their ability to prevent surface flutter. FCS classifications are independent of the methods used for their mechanization. Flight Control systems may consist of more than one subsystem which are not classified herein. FCS have traditionally employed a combination of hardware components consisting of mechanical, hydraulic, pneumatic, electrical and electronic components suitably arranged and programmed to transmit surface commands and to provide feedback of surface and aircraft response as required. Actual hardware component selections are to be limited only by what is available and by what can be developed and shown to be suitable. FCS can be designed to be completely independent from each other or can utilize another FCS component in performing their particular function.

"1.2.1.1 Manual Flight Control Systems (MFCS). MFCS are those using pilot commands as the primary action to initiate control system activity to provide changes in control forces and moments necessary to produce changes in airspeed, altitude, heading, attitude and flight path. MFCS functions include pitch, roll, yaw, side force, lift, drag, trim and thrust.

"1.2.1.2 Aerodynamic Enhancement Flight Control Systems (AEFCS). AEFCS are those systems which improve ride qualities, improve stability of the aircraft or augment the pilot's ability to control.

"1.2.1.3 Automatic Flight Control System (AFCS). AFCS are those systems providing automatic maintenance of or diversion from established flight path condition and/or providing dedicated displays for pilot primary control of the flight path or for monitoring automatic control. AFCS provides automatic activity primarily independent of pilot commands except as required for control wheel steering or to activate, deactivate, preselect or reselect modes of operation. AFCS provide automatic control of such preselected flight conditions as airspeed, altitude, attitude and heading. AFCS may also provide automatic flight path control such as terrain following and precision course direction (auto land and auto nav). The AFCS includes autopilots, autothrottles,

flight directors (including flight instruments), and similar control subsystems.

"1.2.1.4 Limiting Flight Control Systems (LFCS). LFCS are those FCS which provide structural load alleviation or flutter suppression. These controls may act automatically to reduce the combined loads effects of maneuvering when encountering external disturbances (gusts and turbulence). Also they may provide fixed or varying degrees of aerodynamic damping necessary to assure overall flutter-free operation."

Requirement

1.2.2 FCS Operational State Classifications

1.2.2.1 Operational State I (Normal Operation). Operational State I is the normal state of flight control system performance, safety and reliability. This state satisfies MIL-F-8785 or MIL-F-83300 Level 1 flying qualities requirements within the operational flight envelope and Level 2 within the service envelope and the stated requirements outside of these envelopes.

1.2.2.2 Operational State II (Restricted Operation). Operational State II is the state of less than normal equipment operation or performance which involves degradation or failure of only a noncritical portion of the overall flight control system. A moderate increase in crew workload and degradation in mission effectiveness may result from a limited selection or normally operating FCS modes available for use; however, the intended mission may be accomplished. This state satisfies at least MIL-F-8785 or MIL-F-83300 Level 2 flying qualities requirements within the operational flight envelope and Level 3 within the service envelope.

1.2.2.3 Operational State III (Minimum Safe Operation). Operational State III is the state of degraded flight control system performance, safety or reliability which permits safe termination of precision tracking or maneuvering tasks and safe cruise, descent, and landing at the destination of original intent or alternate but where pilot workload is excessive or mission effectiveness is inadequate. Phases of the intended mission involving precision tracking or maneuvering cannot be completed satisfactorily. This state satisfies at least MIL-F-8785 or MIL-F-83300 Level 3 flying qualities requirements.

1.2.2.4 Operational State IV (Controllable to an Immediate Emergency Landing). Operational State IV is the state of degraded FCS operation at which continued safe flight is not possible; however, sufficient control remains to allow engine restart attempt(s), a controlled descent and immediate emergency landing.

1.2.2.5 Operational State V (Controllable to an Evacuable Flight Condition). Operational State V is the state of degraded FCS operation at which the FCS capability is limited to maneuvers required to reach a flight condition at which crew evacuation may be safely accomplished.

Comparison

This requirement defines the classifications of the Operational States for the FCS and therefore a direct comparison with the C-5A cannot be made.

Discussion

The definitions for Operational States I through V are applicable to present and future transport aircraft, but a change is recommended to the term "service envelope." Since Paragraph 6.6 refers to MIL-F-8785 for the

definition for "operational flight envelope," it is assumed that the intent was to use MIL-F-8785 for the definition of "service envelope." However, MIL-F-8785 defines "service flight envelope." This requirement should be revised to reference "service flight envelope" and the Paragraph 6.6 should have this term added with a reference to MIL-F-8785 for the definition.

Recommendation

Revise the requirement as follows:

> Change "service envelope" to "service flight envelope" in last sentence of Paragraphs 1.2.2.1 and 1.2.2.2.

Requirement

1.2.3 FCS Criticality Classification

1.2.3.1 Essential. A function is essential if loss of the function results in an unsafe condition or inability to maintain FCS Operational State III.

1.2.3.2 Flight Phase Essential. A function is flight phase essential if loss of the function results in an unsafe condition or inability to maintain FCS Operational State III only during specific flight phases.

1.2.3.3 Noncritical. A function is noncritical if loss of the function does not affect flight safety or result in control capability below that required for FCS Operational State III.

Comparison

The C-5A has multiple control surfaces controlled through independent, but interconnected flight control subsystems in the pitch, roll and yaw axes. The loss of any independent subsystem by any means does not degrade the FCS below Operational State III. Table 2 shows the degradation and worst Operational States resulting from loss of each independent subsystem regardless of the type or number of failures required to lose the subsystem.

In order to fully evaluate this requirement the term "function" must be defined. It cannot be defined for example as "control of an axis" since there would then be no valid FCS criticality classification other than essential. Obviously this is not the intent. Neither can "function" be satisfactorily applied when defined as "an independent subsystem" since a portion of an FCS does frequently play a role in more than one particular control duty. For example, an elevator FCS may include pitch maneuver control, stability augmentation, trim, autopilot, load alleviation, etc. What emerges as a workable definition is one which can be applied to requirements generally and which is in agreement with FAA interpretations for commercial aircraft. Therefore, "function" will be defined as follows and used as such in the remaining validations. It is also recommended for incorporation into Section 6.6.

Definition - Function: A control function is a particular service or special duty which is performed by any portion of the FCS. Any portion of the FCS may perform more than one function.

Examples of C-5A flight control functions fitting this definition and appropriate criticality classifications are contained in Table 3. This definition is independent of the system or component redundancy or the numbers and types of failures which cause loss of any particular function. Discussions of the C-5A roll, yaw, and pitch axis functions are presented here to provide a basis for validating the requirements of 1.2.3 FCS criticality classification and its subparagraphs.

TABLE 2

C-5A FLIGHT CONTROL (FC) SUBSYSTEMS AND OPERATIONAL STATE
*PER REQUIREMENTS OF 1.2.2

o = Flying Qualities per Operational State I

FC Subsystem	Degradation	Oper State (After Loss)*	
Pitch Axis Subsystems:			
Inboard Elevators	Reduced Pitch Cont. Authority	II	
Outboard Elevators	Reduced Pitch Cont. Authority	II	
Manual Trim	Increased Pilot Work	III	
Stab. Aug.	Moderate Increase Pilot Work	II	
ALDCS	Increased Structural Fatigue and workload	II	
Variable Feel	Increased Pilot Attention	III	
Stallimiter	Increased Pilot Attention	II	o
Auto Pilot	Increased Pilot Work	II	
Roll Axis Subsystems:			
Ailerons	Ruduced Roll Cont. Authority	II	
Flight Spoilers	Reduced Roll Cont. Authority	II	
Manual Trim	Slight Increased Pilot Work	II	
Stab. Aug.	Moderate Increased Pilot Work	II	
ALDCS	Increased Structural Fatigue	II	o
Mech. Feel	Increased Pilot Attention	II	
Auto Pilot	Increased Pilot Work	II	
Yaw Axis Subsystems:			
Upper Rudder	Reduced Yaw Cont. Authority	II	
Lower Rudder	Reduced Yaw Cont. Authority	II	
Manual Trim	None - Use Yaw Aug. Man. Trim	II	o
Yaw Aug. Man. Trim	None - Use Manual Trim	II	o
Stab. Aug.	Moderate Increase Pilot Work	II	
Rudder Limiter	Increased Pilot Attention	II	
Mech. Feel	Increased Pilot Attention	II	
Auto Pilot	Increased Pilot Work	II	
Lift/Drag Subsystems:			
Ground Spoilers	Increased Landing Distance	III	
L.E. Slats (Up)	Increased Landing Distance	III	
(Dn)	Increased Drag; Reduced Speed	III	
T.E. Flaps (Up)	Increased Landing Distance	III	
(Dn)	Increased Drag; Reduced Speed	III	
Throttles	Thrust Control		
Auto-throttles	Thrust Control		

TABLE 3

C-5A FLIGHT CONTROL (FC) FUNCTIONS AND CRITICALITY

Flight Control Function	Surfaces/Devices and Subsystems	Function Criticality*
MFCS:		
Maneuver	All Elev., Var. Feel, and Elevator Manual Controls	1.2.3.1
	Ailerons, Spoilers, and Aileron Manual Controls	1.2.3.1
	Both Rudders and Rudder Manual Controls	1.2.3.2
Trim	Horizontal Stab. (Normal and Emer.)	1.2.3.3
	Ailerons and Trim Controls	1.2.3.3
	Both Rudders and Trim Controls (Norm.)	1.2.3.3
	Both Rudders, Yaw Aug. Man. Trim (Emer.)	1.2.3.3
Lift/Drag	T.E. Flaps, L.E. Slats, Ground Spoilers	1.2.3.2
Thrust	Throttles Control System	1.2.3.1
AEFCS:		
Stability Aug.	Inboard Elevators and SAS	1.2.3.3
	Ailerons and SAS	1.2.3.3
	Both Rudders and SAS	1.2.3.3
Stall Warning	Stick Shaker and Audible Warning	1.2.3.3
Stall Limiting	(None)	
AFCS:		
Auto Control	All Elevators, Horiz. Stab. and A/P (Auto Pilot) Systems	1.2.3.3
	Ailerons, Spoilers and A/P System	1.2.3.3
	Both Rudders and A/P System	1.2.3.3
Automatic Thrust	Auto-Throttle System	1.2.3.3
Dedicated Displays	Flight Director and Flight Instruments	1.2.3.3
	Angle of Attack System and Instruments	1.2.3.3
LFCS:		
Load Control	Inboard Elevators, SAS and ALDCS	1.2.3.3
	Ailerons, SAS and ALDCS	1.2.3.3
Limiting	Both Rudders and Rudder Limiter	1.2.3.3

* 1.2.3.1 = Essential

1.2.3.2 = Flight Phase Essential(as revised)

1.2.3.3 = Noncritical

Roll Control for Manual Maneuvering Function: The C-5A manual maneuvering roll control function is provided by ailerons and spoilers which are normally controlled from pilots commands on the control wheels through the mechanical input system to provide simultaneous commands to all involved surface hydraulic servo actuators. Loss of the complete provisions for manual maneuvering roll control requires the simultaneous existence of at least two improbable mechanical failures or the simultaneous loss of all hydraulic fluid from four systems or the loss of all hydraulic power and failure of the Ram Air Turbine to function. In the first instance, (the highly improbable combination of improbable mechanical failures) all other roll axis functions would remain available at the option of the pilot and the aircraft would be in Operational State IV defined in Paragraph 1.2.2.4. In the event of loss of only the manual roll control function, limited roll control can be provided using the rudders and asymmetric thrust. In the instance of complete loss of hydraulic fluid and/or power the loss of all control including manual roll control would occur. It should be noted that the C-5A manual roll control for maneuvering in particular, as well as the FCS in general have the highest degrees of redundancy in their manual input load paths, hydraulic systems and power supplies and control surfaces of any existing large operational transport aircraft. The manual roll maneuvering control can accurately be called "Essential" as defined in Subparagraph 1.2.3.1.

Roll Manual Trim Control Function: Loss of this manual trim function requires at least two electrical failures whose combined occurrence can easily be described as improbable. However, its loss is noncritical as defined in Subparagraph 1.2.3.3 since the only consequence would be a moderate increase in long duration pilot work load (Rim forces $<$ 5 lbs.) when the autopilot was not operating during any flight phase.

Aileron ALDCS, SAS and A/P Functions: The ALDCS functions include elastic mode suppression and maneuver load control. The roll stability augmentation system automatically smooths turn entry and exit, coordinates the turns, improves dutch roll damping and performs spiral mode stabilization. These remaining aileron functions have no affect on C-5A operational limitations and the loss of any one of these functions is noncritical as defined in Paragraph 1.2.3.3.

Complete Yaw Axis: In the event of complete loss of the yaw axis manual maneuver control function both the manual trim control and emergency rudder control functions provide yaw control. In the case of loss of all rudder functions, limited yaw control can be provided by the roll axis and asymmetric thrust. The aircraft will be in FCS Operational State III and that state meets the requirements defined in 1.2.3.2 as recommended to revision to improve applicability. This yaw SAS is considered noncritical and performs yaw damping and turn coordination.

Complete Pitch Axis: The complete loss of the pitch axis manual maneuver control occurs after loss of control over two essentially independent elevator control systems, i.e., the inboard elevator controls and the outboard elevator controls. After such simultaneous loss, pitch control would be provided by the pitch trim function and the C-5A would be in FCS Operational State III or better. Under that condition, this manual elevator control function meets the requirement of Paragraph 1.2.3.2, as revised. This pitch SAS is considered non-critical and performs pitch damping.

The ALDCS system performs maneuver and gust load control, elastic mode suppression and fatigue relief and is considered to be noncritical. The ALDCS uses portions of the aileron and elevator manual systems and portions of the lateral and pitch SAS.

Discussion

When supplemented by the proposed definition of "Function" the requirements of 1.2.3 are acceptable and are applicable to future transport aircraft with the exception of Paragraph 1.2.3.2 Flight Phase Essential which is confusing. The last phrase of that requirement "inability to maintain FCS Operational State III only during specific flight phases" is believed to be in error. It has been applied in our comparisons as if it read "inability to maintain Operational State II during specific flight phases only." This conclusion was reached because the "inability to maintain FCS Operational State II" (Paragraph 1.2.2.2), which permits no consideration of flight phases automatically leads you to the next degradation of operation state which is III. Further, FCS Operational State III permits the inability to satisfactorily complete particular flight phases. Further, if not able to maintain Operational State III then the aircraft is automatically in Operational State IV which permits only controlled descent to emergency landing.

Recommendation

Retain requirements 1.2.3.1 and 1.2.3.3. Revise requirement 1.2.3.2 as follows:

> "1.2.3.2 Flight Phase Essential. A function is flight phase essential if its function is necessary to prevent an unsafe condition or results in inability to maintain FCS Operational State III during specific flight phases. Its loss must permit subsequent safe return to flight phases (conditions) necessary for continued safe flight and normal landing with at least Operational State III capability."

Requirement

2. APPLICABLE DOCUMENTS

2.1 The following documents, of the issue in effect on the date of invitation for bids or request for proposal, form a part of this specification to the extent specified herein. The requirements of this specification shall govern for flight control system design where conflicts exist between this specification and other reference specifications.

For a listing of these documents, refer to MIL-F-9490D.

2.2 Other Publications. The following documents form a part of this specification to the extent specified herein. Unless otherwise indicated, the issue in effect on date of invitation for bids or request for proposal shall apply.

For a listing of these documents, refer to MIL-F-9490D.

Comparison

As noted in the various detail validation discussions, the C-5A FCS design used many of these specifications or specification guidelines based on these specifications.

Discussion

Some new specifications included in MIL-F-9490D became applicable after the C-5A contract definition. In some cases the new specification may have superseded an older specification that had been imposed on the C-5A.

In accordance with recommended specification changes, some specification additions and deletions are recommended. The requirements as listed are valid and have been satisfied by the C-5A FCS design to the extent noted in the detail validation discussions. The requirements can be demonstrated and should be specified for all future transport type aircraft to the extent noted in the detail validation discussions.

Recommendation

'evise the applicable documents list as noted below.

Specifications - Military

Add:

MIL-S-3950 - Switch, Toggle, Environmentally Sealed, General Specification For

MIL-S-6743 - Switches, Pushbutton and Limit

MIL-A-8064 - Actuators and Actuating Systems, Aircraft Electro-Mechanical, General Requirements For

Delete:

MIL-M-7969 - Motor, AC, 400 Cycle, 115/200 Volt System, Aircraft, General Specification For

MIL-M-8609 - Motor, DC, 200 Volt System, Aircraft, General Specification For

MIL-G-25561 - Grip Assembly, Controller, Aircraft, Type MC-2

Standards

Add:

MIL-STD-1130 - Connections, Electrical, Solderless Wrapped

Delete:

MIL-STD-454 - Standard General Requirements for Electronic Equipment

Other Publications

FAA Advisory Circular

Add:

FAA Advisory Circular AC-20-57A

3. REQUIREMENTS

3.1 System Requirements. The FCS shall comply with the following requirements.

3.1.1 MFCS Performance Requirements. The MFCS shall comply with applicable general flying quality requirements of MIL-F-8785 or MIL-F-83300 and the special performance requirements of the procurement detail specification.

Comparison

The C-5A was designed to meet MIL-F-8785(ASG) Amendment 4, 17 April 1959, FAR 25 and some special requirements added by the procuring activity.

The C-5A was compared to detailed specification requirements in document AFFDL-TR-75-3 titled Evaluation of the Flying Qualities of MIL-F-8785B(ASG) using the C-5A Airplane.

Discussion

The comments on MIL-F-8785 were discussed in AFFDL-TR-75-3 (MIL-F-83300 not applicable to C-5A). Conclusions and recommendations contained in that report are summarized below.

Conclusions

1. The specification represents a substantial improvement over specifications with respect to requirement definition, format and overall clarity.

2. Generally, the C-5A data compare favorably with the specification except in certain sections where the requirements appear to have been based primarily on medium and light weight airplane data.

3. Based on the C-5A data, the following sections of the specification are far too stringent for Class III airplanes.

    3.3.1.2 - Roll mode (T_R)

    3.3.2.4 - Sideslip excursions

    3.3.4 - Roll control effectiveness

Recommendations

1. Additional data from Class III heavy aircraft be gathered to substantiate or revise the requirements in the following sections.

3.2.1.2	Phugoid stability
3.2.2.1	Short period response
3.2.2.2.1	Control forces in maneuvering flight

3.3.1.1	Lateral directional oscillations (Dutch roll)
3.3.1.2	Roll mode (T_R)
3.3.2.4	Sideslip excursions
3.3.4	Roll control effectiveness
3.4.2.2.1	Resistance to loss of control

Recommendation

Accept paragraph 3.1.1 of MIL-F-9490D on the assumption that conclusions and recommendations for changes to MIL-F-8785 will be accomplished.

Requirement

3.1.2 AFCS Performance Requirements. When the following AFCS functions are used, the following specified performance shall be provided. Unless otherwise specified, these requirements apply in smooth air and include sensor error. Except where otherwise specified, a damping ratio (6.6) of at least 0.3 critical shall be provided for nonstructural AFCS controlled mode responses. Specified damping requirements apply only to the response characteristics for perturbations an order of magnitude greater than the allowable residual oscillation.

Comparison

The C-5A AFCS performance requirements for basic response are given for an upset in pitch or heading of ±5 degrees the aircraft shall return to the reference attitude with the first overshoot not exceeding 20 percent of the initial deviation. The C-5A meets these performance requirements for an upset. It is felt that the C-5A AFCS meets the intent of the MIL-F-9490D requirement.

Discussion

This requirement is valid for present and future aircraft, but confusion could exist with the requirements of Paragraph 3.1.3.6.1 of MIL-F-9490D. Paragraph 3.1.3.6.1 covers the stability margins for all aerodynamically closed loop FCS. A change should be made here to reduce this confusion.

Recommendation

Revise the requirement as follows:

> "3.1.2 AFCS Performance Requirements. When the following AFCS functions are used, the following specified performance shall be provided. Unless otherwise specified, these requirements apply in smooth air and include sensor error. Except where otherwise specified, a damping ratio (6.6) of at least 0.3 critical shall be provided for nonstructural inner closed loop AFCS control modes. Specified damping requirements apply only to the response characteristics for perturbation an order of magnitude greater than the allowable residual oscillation."

Requirement

3.1.2.1 Attitude Hold (Pitch and Roll). Attitudes shall be maintained in smooth air with a static accuracy of ±0.5 degree in pitch attitude (with wings level) and ±1.0 degree in roll attitude with respect to the reference. RMS attitude deviations shall not exceed 5 degrees in pitch or 10 degrees in roll attitude in turbulence at the intensities specified in 3.1.3.7. When using a flight controller (turn knob) the aircraft shall return to a wings level attitude when the turn control is placed in the detent position. Accuracy requirements shall be achieved and maintained within 3 seconds of mode engagement for a 5 degree attitude disturbance for MIL-F-8785 Class IV aircraft, and within 5 seconds for MIL-F-8785 Classes I, II and III aircraft.

Comparison

The C-5A AFCS Attitude Hold Mode required an accuracy of ±1 degree over the entire flight regime up to the maneuvering limits. The aircraft was tested and met the C-5A requirement. It is felt that the C-5A would meet the MIL-F-9490D requirement for attitude hold.

Discussion

The requirement as stated is valid for present and future aircraft. It is felt that leaving the performance during maneuvering flight for the aircraft specification was a good decision since this tolerance is dependent upon the type of aircraft, mission requirements and whether CWS is being used.

Recommendation

Retain the requirement as stated.

Requirement

3.1.2.2 Heading Hold. In smooth air, heading shall be maintained within a static accuracy of ±0.5 degree with respect to the reference. In turbulence, RMS deviations shall not exceed 5 degrees in heading at the intensities specified in 3.1.3.7. When using a flight controller, heading hold shall automatically engage as the controller is returned to the detent position.

Comparison

Heading hold mode specifications for the C-5A AFCS are very similar to the above. Areas of difference include the static accuracy and turbulence requirements. The C-5A requirement for static accuracy was within ±1.0 degree with respect to the reference. The C-5 AFCS requirement for heading hold did not differentiate between smooth and turbulent air. The capability to control tolerances within the AFCS has improved since the mid 60's, making the ±0.5 degree accuracy requirement realizable. The C-5A heading hold mode is the basic mode of the AFCS roll axis with or without the Roll Control Wheel Steering (CWS) mode operating. When the autopilot roll axis is engaged, the heading hold mode is automatically engaged and the roll rate CWS mode is automatically on (armed). The pilot can change the aircraft heading by using the turn knob (controller) or by using rate CWS. When using the rate CWS when the pilot force on the wheel exceeds the preset threshold level, the mode is engaged and the aircraft will roll at a rate depending upon pilot force. If the pilot releases his force below the threshold level and the aircraft is above 7 degrees bank angle, the aircraft will be automatically held at that bank angle. If the pilot releases his force below the threshold level when the bank angle is less than 7 degrees, the aircraft will be automatically in the heading hold mode. The heading existing at the time the force falls below this threshold will be the new reference heading. When the turn knob is used, the mode functions in a similar manner as CWS. The amount of rotation of the knob will command a specific roll angle as long as the knob is left in that position. When the turn knob is returned to detent, the aircraft will roll to wings level and the heading existing at the time the aircraft is at 7 degrees bank angle, is the heading that will be held.

The C-5A heading hold mode was originally configured like the MIL-F-9490D requirement where the heading existing at the instant the controller returned to the detent position was the heading which was held. During flight testing of that C-5A mode, when the aircraft was in a 30 degree bank turn and the controller was returned to detent, the aircraft would roll out at the turn and then roll into a turn in the opposite direction to go to the heading existing at the instant the detent was reached. The pilots objected to this implementation and suggested that a change be made. The heading hold mode is normally used to change aircraft direction and not for achieving a specific heading. If the pilot wishes to go to a specific heading, he uses the heading select mode.

Discussion

It may be arguable that a heading hold accuracy of ±0.5 degrees does not

appreciably enhance mission effectiveness or aircraft operational efficiency over an accuracy of ± 1.0 degrees for the heading hold mode. Since, however, the state-of-the-art now allows realization of the more stringent requirement without undue penalty in cost, the requirement is considered valid.

The 5 degree RMS heading deviation requirement for operation in light turbulence is desirable. This prevents design of an easily saturable mode while not restricting the functional design of the overall AFCS.

The problem with the heading hold mode experienced by the C-5A would likely be encountered with other large aircraft. A change is recommended to allow for engagement of the mode using another parameter along with the detent position to indicate the reference heading.

The requirement states that heading hold shall automatically engage as the controller is returned to the detent. The use of the word "as" makes this confusing. The word "when" is proper in this case. A majority of the aircraft use the detent position as the logic for going to the heading hold mode.

Recommendation

Revise the requirement as follows.

Change last sentence to:

"When using a flight controller, heading hold shall automatically engage when the controller is returned to the detent position."

Add the following sentence:

"In the event the aircraft turns in the opposite direction to hold the referenced heading when the controller is returned to detent, another parameter may be used in addition to the detent to engage the heading hold mode as the aircraft approaches level flight."

Additional Data (For "Users' Guide")

The sentences indicated by the left vertical sideline should be added to the background information and "Users' Guide."

Requirement

3.1.2.3 Heading Select. The aircraft shall automatically turn through the smallest angle to any heading selected or preselected by the pilot and maintain that heading to the tolerances specified for heading hold. The contractor shall determine a bank angle limit which provides a satisfactory turn rate and precludes impending stall. The heading selector shall have 360 degrees control. The aircraft shall not overshoot the selected heading by more than 1.5 degrees with flaps up or 2.5 degrees with flaps down. The roll rate shall not exceed 10 deg/sec and roll acceleration shall not exceed 5 deg/sec/sec for MIL-F-8785 Classes I, II and III aircraft, or double these values for MIL-F-8785 Class IV aircraft.

Comparison

The C-5A requirements for the heading select mode compare closely with MIL-F-9490D. However, instead of specific limits on roll rate and roll acceleration, the C-5 requirements state that entry into, and termination of, the turn shall be smooth and rapid. Additionally, the C-5 specification limited heading overshoot to 1.5 degrees but did not distinguish between flaps up or down operation. The limit of bank angle while turning to the selected heading was specified as 30° for the C-5. The C-5A automatic heading select mode is interlocked with the navigation control panel and is obtained by selecting HDR NAV on the AFCS panel and HDG on the navigation control panel.

Discussion

C&I specification compliance testing was accomplished during Category II evaluation of the C-5A AFCS at AFFTC, Edwards AFB. The mode was evaluated by commanding heading changes of 5, 10 and 60 degrees, both left and right, using the heading selector on the HSI. A heading change of 175 degrees was also performed to verify that the A/P maneuvered the aircraft through the smallest angular distance to the new heading. The A/P smoothly controlled the aircraft during roll in and roll out with no appreciable (less than 2°) overshoot of the selected heading. On-hand flight test response plots do not have a record of roll rate or roll acceleration for comparison with new limits. The maximum bank angle command was 27 degrees. When a 175° heading change was selected, the A/P hestiated momentarily before commanding a slow roll into the turn. In all cases the aircraft was maneuvered through the smallest angular distance to the new heading.

The imposition of limits on roll rate and roll acceleration when maneuvering to the new heading is seen as a relaxation of the qualitative "smooth and rapid" requirement of the C-5 AFCS. It establishes an upper limit for the rates and accelerations but does not address a minimum acceptable.

Recommendation

Insert the following between the last two sentences: "Entry into and termination of the turn shall be smooth and rapid."

Requirement

3.1.2.4 Lateral Acceleration and Sideslip Limits. Except for flight phases using direct side force control, the following performance shall be provided whenever any lateral-directional AFCS function is engaged. Lateral acceleration refers to apparent (measured, sensed) body axis acceleration at the aircraft center of gravity.

3.1.2.4.1 Coordination in Steady Banked Turns. Sideslip angle shall not be greater than 2 degrees and lateral acceleration shall not exceed 0.03g, while at steady bank angles up to the maneuver bank angle limit reached during normal maneuvers with the AFCS engaged. For rotary wing aircraft, only the lateral acceleration limit applies.

Comparison

The C-5A AFCS provides automatic turn coordination whenever the yaw and lateral augmentation systems or the autopilot roll and yaw axes are engaged. The performance requirements were identical to those specified above; i.e., 2 degree sideslip angle limit and 0.03g lateral acceleration limit. The maximum maneuver bank angle is 35 degrees when using the turn knob or control wheel steering and 27 degrees for other roll autopilot modes. A bank angle limit of 7.5 degrees is imposed after glideslope capture when operating in localizer mode or when in the VOR or TACAN track modes. Validation tests were accomplished at AFFTC at altitudes of 35,000, 25,000 and 10,000 feet (the 10,000 foot evaluation was accomplished for both approach and landing flap configurations). After trimming the aircraft, straight and level, at appropriate cruise, approach, or landing airspeed, the autopilot was engaged with altitude-hold. Rapid turns to the right and left were performed using the turn knob. The turns were continued thru 180 degrees heading change with roll out accomplished as rapidly as possible. The turn knob was then used to roll rapidly from 30 degrees right to 30 degrees left without hesitating at wings-level. The maneuver was then repeated, rolling the opposite direction.

During the 180 degree heading changes, altitude increases between 300 and 1,200 feet were experienced. Airspeed decreased between zero and 17 KIAS. The aircraft response was smooth and well coordinated throughout the maneuver. The crew experienced no feeling of sideslip or lateral acceleration.

Discussion

While the C-5A AFCS experience validates this requirement, a look at the application of such a requirement for future transports must be taken. At the present time Lockheed-Georgia is defining several improved C-130 configurations. Wind tunnel data taken on one such model show asymmetric Cy and Cn vs. β as shown on Figure 1 (3.1.2.4.1). The implication of this data is that to maintain the aircraft in trim a crab angle must be established to balance the unsymmetrical aerodynamic forces. It is also possible to envision the necessity to establish a trim bank angle reference other than 0°. These unsymmetric aerodynamic forces are common for turboprop aircraft.

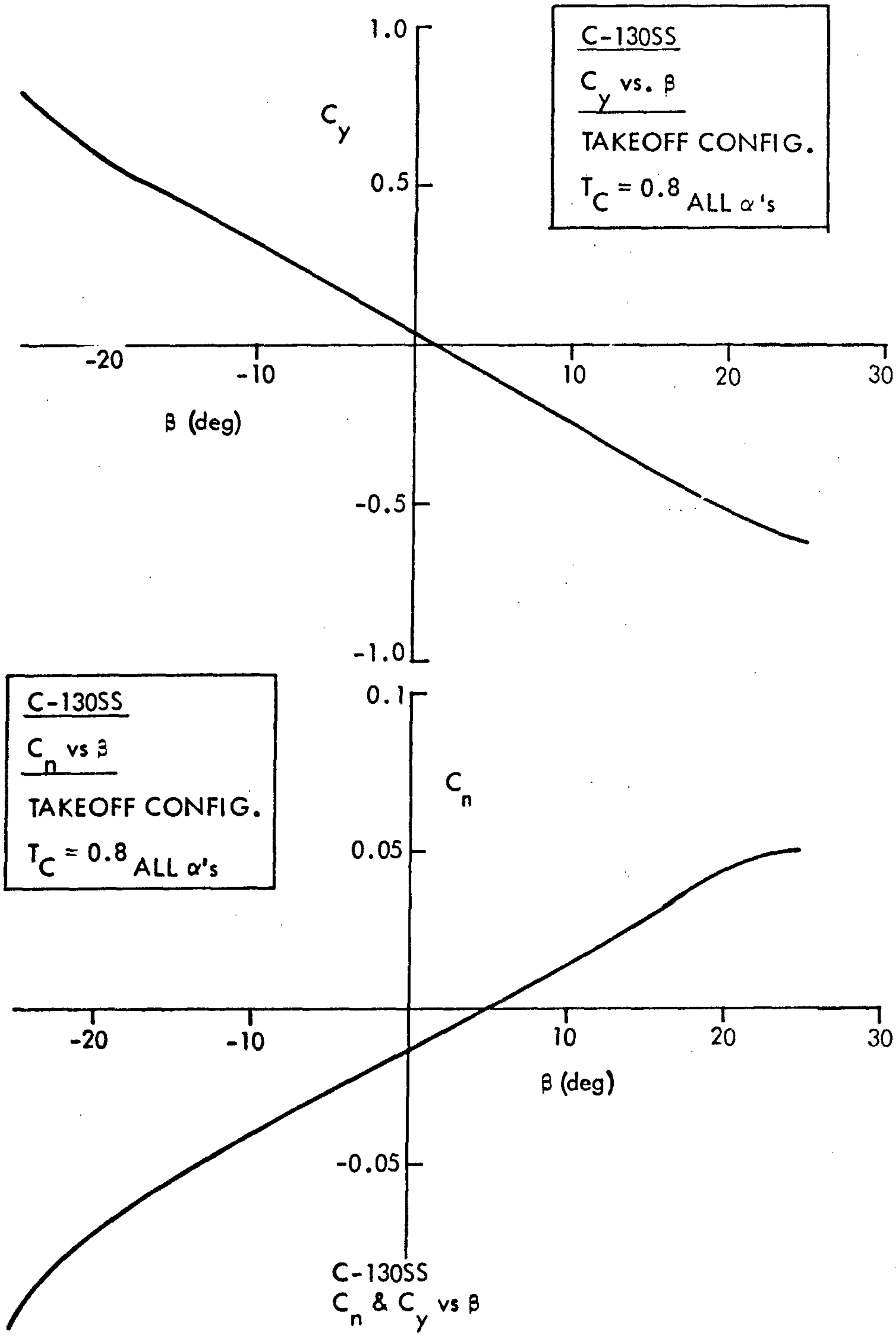


FIGURE NO. 1 (3.1.2.4.1). C-130SS C_y vs. β and C_n vs. β

For these aircraft it is necessary to remove the absolute sideslip angle restriction from steady banked turns. A restriction is proper, but it should be stated in terms of incremental sideslip from unaccelerated flight reference sideslip value.

Recommendation

Change the first sentence to read:

> "The incremental sideslip angle shall not exceed 2 degrees from the trimmed value, and lateral acceleration shall not exceed 0.03g, while at steady bank angles up to the maneuver bank angle limit reached during normal maneuvers with the AFCS engaged."

Requirement

3.1.2.4.2 Lateral Acceleration Limits, Rolling. Body axis lateral acceleration at the cg shall not exceed ±0.1g for aircraft with roll rate capability up to 30 deg/sec, ±0.2g for aircraft with roll rate capability of 30 to 90 deg/sec, or ±0.5g for aircraft with roll rates over 90 deg/sec. These limits shall be satisfied for aircraft in essentially constant altitude flight while rolling smoothly from one side to the other at bank rates up to the maximum obtainable through AFCS modes.

Comparison

The C-5A AFCS was limited to ±0.1g lateral acceleration when rolling under the conditions specified above. Since the maximum roll rate capability of the AFCS is less than 30 degrees per second, it is compatible with the above paragraph. Verification testing was accomplished qualitatively.

Discussion

The requirement is considered valid, both for the C-5A and for future transport aircraft. It created no unusual problems during design or test and is acceptable as it stands.

Recommendation

Accept the paragraph "as is."

Requirement

3.1.2.4.3 Coordination in Straight and Level Flight. The accuracy while the aircraft is in straight and level flight shall be maintained within a sideslip angle of ±1 degree and a lateral acceleration of ±0.02g at the cg, whichever is lower. For rotary wing aircraft, only the lateral acceleration limit applies.

Comparison

The C-5A AFCS provides automatic turn coordination as functions of the yaw and lateral augmentation systems. The performance requirements for coordination in straight and level flight were not specified in the detail C-5A flight controls specification. A review of the available data reveals that the C-5A AFCS could meet this requirement.

Discussion

The discussion of Paragraph 3.1.2.4.1 applies to the above paragraph as well.

Recommendation

Change the first sentence to read:

> "The accuracy while the aircraft is in straight and level flight shall be maintained within an incremental sideslip angle of ±1 degree from the trimmed value and a lateral acceleration of ±0.02g at the cg, whichever is lower."

Requirement

3.1.2.5 Altitude Hold. Engagement of the altitude hold function at rates of climb or descent less than 2,000 fpm shall select the existing indicated barometric altitude and control the aircraft to this altitude as a reference. The resulting normal acceleration shall not exceed 0.2g incremental for MIL-F-8785 Classes I, II and III aircraft, or 0.5g incremental for MIL-F-8785 Class IV aircraft. For engagement at rates above 2,000 feet per minute the AFCS shall not cause any unsafe maneuvers. Within the aircraft thrust-drag capability and at steady bank angles, the mode shall provide control accuracies specified in Table I.

TABLE I. MINIMUM ACCEPTABLE CONTROL ACCURACY

BANK ANGLE (DEG.) / ALT. (FT.)	0 – 1	1 – 30	30 – 60
55,000 to 80,000	±0.1% at 55,000 varying linearly to ±0.2% at 80,000	±60 ft. or ±0.3% whichever is larger	±90 ft. or ±0.4% whichever is larger
30,000 to 55,000	±0.1%		
0 to 30,000	±30 ft.		

These accuracy requirements apply for airspeeds up to Mach 1.0. Double these values are permitted above Mach 1.0 and triple these values apply above Mach 2.0. Following engagement or perturbation of this mode at 2,000 feet per minute or less, the specified accuracy shall be achieved within 30 seconds. Any periodic residual oscillation within these limits shall have a period of at least 20 seconds.

Comparison

The C-5A Altitude Hold Mode performance requirements were the same as given in Table 1 for the a flight envelope up to 40,000 feet in altitude. Engagement at rates of climb and descent up to 2,000 fpm were tested and found to be acceptable. The C-5A did not have any maximum performance requirement on normal acceleration when engaging at 2,000 fpm. It is felt that the C-5A can meet this requirement.

Discussion

This requirement is reasonable and applicable to present and future aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.1.2.6 Mach Hold. The Mach number existing at the engagement of Mach hold shall be the reference. After engagement and stabilization on Mach hold, the AFCS shall maintain indicated Mach number and the error shall not exceed ±0.01 Mach or ±2 percent of indicated Mach, whichever is larger, with respect to the reference. Any periodic oscillation within these limits shall have a period of at least 20 seconds. The contractor shall establish a mode response or maximum time to capture requirement which is suitable for the mission phase.

Comparison

The C-5A AFCS contains Mach hold and adjust mode on pitch as well as an automatic throttle Mach hold mode. The C-5A Mach hold mode requires the referenced Mach number be held to ±.015 between 0.60 and 0.785 Mach. The pilot has provisions for varying the engaged reference between 0.60 and 0.785. It is felt that the C-5A meets the intent of this requirement.

Discussion

This requirement is applicable to a Mach hold mode using either the autopilot pitch axis or an automatic throttle system. The RFP and the FCS specification should define which is to be used. Lockheed's experience on installing automatic throttle systems on the QB-47, C-141 and C-5A has shown that some adjustment capability must be made available for the pilot. It is very difficult to engage this mode at the Mach number required in adverse weather. ARINC Characteristic No. 558 (Air Transport Automatic Throttle System) indicates a full range of adjustment for their system. It is recommended that this MIL-F-9490D requirement contain an adjustment requirement.

Recommendation

Revise the requirement as follows. Add the following sentence:

> "Adjustment capability of at least ±0.01 Mach shall be available to allow the pilot to vary the reference Mach number around the engaged Mach No."

Requirement

3.1.2.7 Airspeed Hold. The airspeed existing at the engagement of airspeed hold shall be the reference. Indicated airspeed shall be maintained within ±5 knots or ±2 percent, whichever is greater, of the reference airspeed. Any periodic oscillation within this limit shall have a period of at least 20 seconds. The contractor shall establish a mode response or maximum time to capture requirement which is suitable for the mission phase.

Comparison

The C-5A AFCS contains an airspeed hold and adjust mode on pitch as well as an automatic throttle airspeed hold mode. The C-5A airspeed on pitch mode required the airspeed be held to ±4 knots calibrated air speed from the referenced airspeed on the vertical scale flight instruments and with provisions of varying the engaged airspeed. The C-5A Automatic Throttle System airspeed hold mode had the same performance requirement. The C-5A meets the intent of this requirement.

Discussion

This requirement is applicable to an airspeed hold mode using either the autopilot pitch axis or an automatic throttle system. The RFP and the FCS specification should define which is to be used. Lockheed's experience on installing automatic throttle systems on the QB-47, C-141 and C-5A has shown that some adjustment capability must be available for the pilot. It is very difficult to engage the mode at the control airspeed required in adverse weather. ARINC Characteristic No. 558 (Air Transport Automatic Throttle System) indicates a full range of adjustment for their system. It is recommended that this MIL-F-9490D requirement contain an adjustment requirement.

Recommendation

Revise the requirement as follows. Add the following sentence:

> "Adjustment capability of at least ±10 knots shall be available to allow the pilot to vary the reference airspeed around the engaged airspeed."

Requirement

3.1.2.8 Automatic Navigation

3.1.2.8.1 VOR/TACAN. When preconditions for radial capture are satisfied the AFCS shall cause the aircraft to maneuver to acquire the radial beam center. Maximum roll rate and attitude commands shall be limited to provide a smooth capture and subsequent tracking of the radial. The following performance requirements for VOR are stated in terms of crosstrack error (feet) and radial error (expressed in u amps; 1 degree = 15 u amps) to provide for systems using either ARINC 547 or 579 VOR receivers. For ARINC 547 receivers, only the radial error applies. Crosstrack error applied to the ARINC 579 receiver operating in the primary mode (co-located VOR/DME), and radial error applies in the reversionary mode (DME inoperative or not available).

Comparison

The C-5A CEI specification for the VOR/TACAN navigation modes of the C-5A airplane requires that bank angle not exceed 30 degrees during intercept, but has no requirement for roll rate limit. The C-5A mechanization of both VOR and TACAN modes does include a rate limit and position limit on roll angle, as well as a 45° course cut limit.

Discussion

This paragraph covers only general requirements for VOR and TACAN navigation modes and definition of terms. It is reasonable and acceptable.

Recommendation

Accept "as is."

Requirement

3.1.2.8.1.1 VOR Capture and Tracking. Overshoot shall not exceed 5,800 feet (20 ua) beyond the desired ground track line in a no-wind condition for captures 50 miles or more from the station with intercept angles up to 45 degrees. Following capture at 50 miles or more, the aircraft shall remain within an average of 5,800 feet (20 ua) from the VOR radial beam center, with this error allowance decreasing proportional to the distance from the VOR station. Average tracking error shall be measured over a 5 minute period between 50 and 10 miles from the station or averaged over the nominal aircraft flight time between the same distance limits, whichever time is shorter.

Comparison

The C-5A CEI specification requires that there be no more than two overshoots during capture, while this paragraph requires only a maximum overshoot distance. The average tracking error allowance is really an average beam angle error as defined in this paragraph. The C-5A tracking error allowance is 0.1 of full scale course bar deflection, which permits 2.00 degrees of beam error maximum as compared to a 1.26 degree maximum average error which is equivalent to 5800 feet at 50 miles.

Discussion

The use of the term "average error" is objectionable since large "hunting" errors could occur to right and left of the beam and still result in a small "average" error.

Recommendation

It is recommended that "root mean square" be substituted for "average" in the last sentence.

Requirement

3.1.2.8.1.2 TACAN Capture and Tracking. Overshoot shall not exceed 6,300 feet beyond the desired ground track line in a no-wind condition for captures 120 miles or more from the station with intercept angles up to 30 degrees. The required 0.3 damping ratio shall be exhibited for continuous tracking between 120 miles and 20 miles from the station.

Comparison

The C-5A is required to capture with no more than two overshoots and to maintain track within 10% of full scale course bar deflection. This amounts to 2 degrees of beam error.

Discussion

There is no TACAN tracking accuracy requirement in this paragraph.

Recommendation

It is recommended that the following sentence be added to the end of this paragraph:

> "Following capture at 120 miles or more, the aircraft shall remain within a root mean square of 6,300 feet from the TACAN beam center, with this error allowance decreasing proportional to the distance from the TACAN transmitter."

Requirement

3.1.2.8.1.3 Overstation. The VOR/TACAN mode shall include automatic means for maintaining the aircraft within ±1 degree of aircraft heading or ground track existing at the inbound edge of the VOR zone of confusion (ZOC). During overflight of the ZOC, adjustment of the preset course heading or its equivalent shall cause the roll AFCS to maneuver the aircraft to capture the appropriate outbound radial upon exiting from the ZOC. The VOR/TACAN capture maneuvering limits may be reinstated during overstation operation.

Comparison

The C-5A is required to maintain compass heading within ± 2 degrees through the VOR zone of confusion while bank angle transients must be less than 5 degrees. Otherwise, the C-5A meets the requirements of this paragraph.

Discussion

The C-5A requirements are intended to apply in the presence of atmospheric disturbances as specified in MIL-F-8785B.

The implication in this paragraph is that the requirement is meant to apply only in the no-wind condition, as stated in previous paragraphs.

Recommendation

It is recommended that the following be added at the end of the first sentence: "in a no-wind condition."

Requirement

3.1.2.9 Automatic Instrument Low Approach System. The approach mode of the AFCS shall respond to localizer signals for lateral guidance and glide slope signals for vertical guidance. The system shall be designed to automatically steer the aircraft to a minimum decision height of 100 feet during ICAO Category II weather minimums. The system shall provide timely warning to permit the pilot to complete the landing if runway visual contact is established or to safely execute a go-around following any single failure or combination of failures not shown to be extremely remote as defined in 6.6. The system shall comply with the tracking requirements of 3.1.2.9.1 through 3.1.2.9.3 for probable combinations of headwinds to 25 knots, tailwinds to 10 knots, and crosswinds to 15 knots, with the probability of occurrence of such winds and associated turbulence and wind shears as specified in 3.1.3.7.3.

Comparison

The C-5A AFCS contains an automatic approach mode which automatically steers the aircraft to a minimum decision height of 100 feet using a Category II localizer and glide slope ground installation. The mode was designed to meet the requirements of Contract End Item Detail Specification CP 40002-6B, FAR Part 25 and TSO-C9C. The FAA Advisory Circular AC NO:20-57A was used as a guideline in evaluation of the mode. This mode on the C-5A is completely monitored when used with the autoland mode and will automatically disengage the affected axis when a failure occurs and will illuminate warning lights in front of the pilot and copilot. When this mode is used independent of the autoland, it is failsafe in that the pilot can safely execute a landing after any single failure or combination of failures which are not remote by utilizing the Flight Director flight instruments. This system was designed for probable combinations of headwinds to 25 knots, tailwinds to 10 knots, and crosswinds to 15 knots and wind shears of 8 knots/100 feet and 4 knots/100 feet.

It is felt that the C-5A meets the intent of this requirement.

Discussion

This requirement is basically a good requirement with the exception of the last sentence which refers to Paragraph 3.1.3.7.3 for the probability of occurrence of such winds and for wind shear. Examination of Paragraph 3.1.3.7.3 finds no probability of occurrence of the winds to be designed. A curve is given which shows the cumulative probability of exceeding given wind speed. This area should be clarified in better terms so that there can be no question as to the requirements.

An attempt was made to utilize the wind shear equation of Paragraph 3.1.3.7.3.2. It is not clear how this equation and the curve of mean wind exceedance are used together to determine mean wind, U.

This area of probability of occurrence and wind shear must be made completely

clear to the user of MIL-F-9490.

The title of this paragraph does not reflect the requirements that are stated.

Recommendation

Change the title of the requirement to "Automatic Approach System."

Perform an investigation into the proper method of describing the probability of occurrence of winds encountered during approach and landing as it relates to reported winds by the tower and describe a clear method of determining design wind shear to be considered. All terms should be defined completely and all data should be in the same units of measure. After this investigation, revise the last sentence of this requirement.

Requirement

3.1.2.9.1 Localizer Mode. The AFCS shall cause the aircraft to maneuver to acquire the localizer beam. Heading or roll rate and attitude commands shall be limited to provide a smooth capture and subsequent tracking of the localizer beam. Overshoot shall not exceed 0.5 degrees (37.5 ua) radial error from localizer beam center for captures with initial intercept angles of 45 degrees at 8 miles from runway threshold and increasing linearly to 60 degrees at 18 miles from runway threshold in a no wind condition. During localizer capture the system shall exhibit a damping ratio of at least 0.1 within the noted capture ranges, including the effects of system nonlinearities. The system shall be considered to be tracking whenever the following conditions are satisfied: localizer beam error is 1 degree (75 ua) or less, localizer beam rate is 0.025 deg/sec (2 ua/sec) or less, and roll attitude is 5 degrees or less. During beam tracking the system shall exhibit a damping ratio of 0.2 or greater at a distance of 40,000 feet from the localizer transmitter. The AFCS shall maintain the aircraft 2σ position within 0.33 degrees (25 ua) of localizer beam center whenever the aircraft is between (1) 40,000 feet horizontal distance from the localizer transmitter, and (2) the point where 100 feet above the ground is reached; these criteria shall be based on a Category II localizer ground installation and 10,000 foot runway is defined by ICAO Annex 10.

Comparison

The C-5A localizer mode requirements are as follows:

> Localizer - The AFCS shall maintain constant heading until the air vehicle is within ±150 microamperes of the beam center, at which point the air vehicle shall proceed to capture the beam. Beam entry shall be smooth, with the beam intercept angles of 45 degrees at 8 miles out increasing linearly to 60 degrees at 18 miles out. The initial overshoot, during capture, shall not exceed 75 microamperes. There shall be no more than two overshoots. The second overshoot shall not exceed 30 microamperes. When tracking the beam, the damping factor of the tracking mode shall be greater than 0.3. When subjected to a wind shear of 8 knots/100 feet of altitude, starting at any time during the approach and continuing for 300 feet of altitude, the steady-state deviation from the indicated beam center line shall not exceed 40 feet, and the maximum deviation shall not exceed 80 feet. Any steady-state oscillations shall have a period greater than 10 seconds and an amplitude less than ±15 microamperes. The transient errors shall not exceed 30 microamperes.

The C-5A localizer mode was evaluated and found to meet the C-5A requirements. The overshoot at an intercept angle of 45 degrees was less than 15 microamperes. The maximum deviation from the localizer beam center during approaches from 800 feet altitude point averaged 12 microamperes. The 2 sigma deviation was 10 microamperes throughout all approaches.

The MIL-F-9490D requirements are stated in different terms, but it is felt that the C-5A AFCS localizer mode would meet these requirements.

Discussion

It is felt that the requirements of this paragraph are too stringent and do not provide maximum designer freedom while retaining required flight safety.

The overshoot requirement of 0.5 degrees (37.5 microamperes) radial error is very tight and could require a special design such as a variable gain system for a requirement that is not critical. The point at which the beam capture is initiated should be specified. It is felt that 150 microamperes is the best point to start beam capture. This requirement states that a damping ratio of 0.2 or greater shall be exhibited during the track mode at a distance of 40,000 feet from the transmitter. This does not give the required damping before and after the 40,000 foot point. This damping ratio should be required throughout the tracking mode. The tracking accuracy of the requirement is more stringent than the FAA Category II approach requirement of Advisory Circular AC No. 120-129. It is felt that the FAA requirements should be used since these are the requirements that are generally acceptable to the industry.

Recommendation

Revise the paragraph as follows:

"3.1.2.9.1 Localizer Mode. The AFCS shall maintain a constant heading until the aircraft is within ±150 microamperes of the beam center, at which point the aircraft will be maneuvered to capture the localizer beam. Heading or roll rate and attitude commands shall be limited to provide a smooth capture and subsequent tracking of the localizer beam. The initial overshoot, during capture, shall not exceed 75 microamperes and the system shall exhibit a damping ratio of at least 0.1 with intercept angles of 45 degrees at 8 miles from runway threshold and increasing linearly to 60 degrees at 18 miles from runway threshold in a no wind condition. For intercept angles less than 45 degrees, the FCS shall always maneuver the aircraft toward the runway. There shall be no movement away from the runway threshold during capture. The system shall be considered to be in the tracking mode whenever the following conditions are satisfied: localizer beam error is 1 degree (75μa) or less, localizer beam rate is 0.025 deg/sec (2μa/sec) or less. During beam tracking the system shall exhibit a damping ratio of 0.2 or greater. From the outer marker to an altitude of 300 feet above runway elevation on the approach path, the AFCS shall maintain the aircraft 2 sigma position within 0.47 degrees (35μa) of the localizer beam center. From the 300 feet above runway elevation on the approach path to the decision altitude of 100 feet, the AFCS shall maintain the aircraft 2 sigma position within 0.33 degrees (25μa). The performance during the tracking mode shall be free of sustained oscillations. These criteria shall be based on a Category II localizer ground installation and 10,000 feet runway as defined by ICAO Annex 10.

Requirement

3.1.2.9.2 Glide Slope Mode. The pitch AFCS shall cause the aircraft to maneuver to acquire the glide slope beam. Neither the position of the aircraft above or below the glide slope nor vertical speed of the aircraft at time of mode selection shall be incorporated as a precondition for mode engagement. When preconditions are satisfied, overshoot shall not exceed 0.16 degrees (35 ua) of radial error from glide slope beam center when capturing from below the beam in level flight at an altitude greater than 800 feet above the glide slope transmitter datum altitude in a no-wind condition. The system shall exhibit a damping ratio of 0.085 or greater subsequent to the first overshoot for the conditions defined. On a Category II ILS ground facility (including 10,000 foot runway) as defined in ICAO Annex 10, the pitch AFCS shall maintain the aircraft glide slope antenna 2 o opposition within 0.16 degrees (35 ua) of beam center or within 12 feet of beam center, whichever is greater, between the altitudes of 700 feet and 100 feet above the glide slope transmitter datum.

Comparison

The requirement of the C-5A AFCS glide slope mode are given below:

> Glideslope - During bracketing of the glideslope, the initial overshoot shall be less than 30 microamperes. The second overshoot shall not exceed 30 microamperes. The transient errors shall not exceed 30 microamperes. The error shall remain less than 10 microamperes from 30 seconds after glideslope engagement. The damping factor of the glideslope tracking mode after an initial displacement from the beam shall be greater than 0.35. When subjected to a fore-aft wing shear of 4 knots/100 feet, beginning at 500 feet, the vertical deviation from the ideal glideslope shall not exceed ± 4 feet at the runway threshold, assuming constant airspeed. When the airspeed is allowed to vary ± 4 knots from the nominal value, and no wind shear is present, the vertical deviation from the ideal glideslope at the runway threshold shall also not exceed ± 4 feet.

The C-5A requirements are more stringent than the MIL-F-9490D requirement; therefore, it is felt that the C-5A meets the intent of this requirement.

Discussion

It is felt that this is a good requirement, but some changes are required. Capture performance requirements are only given for captures from below the beam. At the present time, more and more approaches are being made at a steeper angle due to environmental (noise) considerations; therefore, the performance requirements for capture should be given for above and below the beam. This requirement also limits the capture performance requirements to an altitude greater than 800 feet above the glideslope transmitter datum altitude. The capture requirements should be met at any point of capture.

The damping ratio requirement of 0.085 or greater after the first overshoot is not acceptable. This low of a damping ratio would be just as bad as neutral stability and could induce PIO (Pilot Induced Oscillation). The damping ratio after the first overshoot should be similar to the localizer mode.

The transient error that could occur during beam tracking should be covered in this requirement. The transient error should never exceed the error allowed for the first overshoot.

The 2 sigma tracking requirements of 0.16 degrees (35μa) or $\pm$12 feet of beam center is felt to be reasonable. This tracking accuracy is the same as that required in Advisory Circular AC 120-29.

Recommendation

Revise the requirement as follows:

> "3.1.2.9.2 Glideslope Mode. The pitch AFCS shall cause the aircraft to maneuver to acquire the glideslope beam. Neither the position of the aircraft above or below the glide nor vertical speed of the aircraft at time of mode selection shall be incorporated as a precondition for mode engagement. When preconditions are satisfied, the first overshoot shall not exceed 0.16 degrees (35μa) of radial error from glideslope beam center when capturing in a no wind condition from above or below the beam under normal approach configurations. The system shall exhibit a damping ratio of 0.20 or greater subsequent to the first overshoot and the transient errors encountered during the tracking mode shall not exceed 0.16 degrees (35μa) of radial error from glideslope beam center. When using a Category II ILS ground facility (including 10,000 foot runway) as defined in ICAO Annex 10, the pitch AFCS shall maintain the aircraft glideslope antenna 2 sigma position within 0.16 degrees (35μa) of beam center or within 12 feet of beam center, whichever is greater, between the altitudes of 700 to 100 feet above the glideslope transmitter datum."

Requirement

3.1.2.9.3 Go-around mode. The automatic go-around mode shall be manually engaged only. The AFCS shall be designed such that no single failure, or combination of failures not extremely remote, will cause the aircraft to maneuver to increase the rate of descent upon engaging the go-around mode. If the go-around mode is designed for concurrent operation with other automatic control systems, a single switch location or pilot action shall engage all systems into the appropriate mode for go-around. Should one or any combination of concurrently operating automatic control systems be inoperative at the time of AFCS go-around mode engagement, the AFCS shall comply with the performance requirements based on normal go-around procedures including manual management of thrust, flaps, and landing gear.

Comparison

The C-5A go-around mode can be engaged by either the pilot or copilot by depressing the go-around switch located on their respective control wheels. When this switch is depressed the autopilot pitch and roll axes are disengaged, the automatic throttles will move the throttles to the high electrical limit, and the proper go-around commands are presented on the flight instruments from the go-around computer through the flight director. The pilot or copilot fly the aircraft manually for the go-around mode. The pilot must manually move the throttles from the automatic throttle electrical limit to the proper position for go-around. The autopilot roll and/or pitch axes can be reengaged and then the aircraft can be controlled during the go-around by use at the control wheel steering (CWS) mode or the pitch and turn controls.

The C-5A has a manual go-around system and therefore cannot comply with the intent of this requirement.

Discussion

The use of an automatic go-around mode would depend on the aircraft and mission requirements. If such a mode is required then this requirement with a slight modification would be relevant for present and future aircraft. The go-around command signals should be displayed on the flight instruments to allow the pilot to monitor the automatic system and to perform a manual go-around in the event of an autopilot failure.

Recommendations

Revise the requirement by adding the following:

> "The go-around commands shall be sent to the flight director system for display on the flight instruments."

Requirement

3.1.2.9.3.1 Pitch AFCS Go-Around. The pitch AFCS shall cause the aircraft to smoothly rotate sufficiently to establish a positive rate of climb such that the aircraft will not intersect the obstacle clearances planes defined in FAA Advisory Circular 120-29 more often than 1 in 10^6 events for the wind conditions specified in 3.1.2.9, and including high altitude, hot day conditions as defined by the procuring activity. In the event of inadvertent loss of an engine just prior to or during automatic go-around, the system shall not cause the aircraft to approach stall within 30 seconds of mode engagement, based on design approach speed. If operating procedures require the mode to be disengaged upon inadvertent loss of an engine, a timely warning shall be provided for the pilot to initiate the disengage procedure. Disengagement under this condition shall be accomplished manually.

Comparison

The C-5A does not contain an automatic go-around mode. The C-5A go-around signals are displayed on the flight instruments for the pilot to use for a manual go-around or utilize the CWS mode of the autopilot. The C-5A contains a completely independent go-around system to insure that after any failure in the autopilot the pilot would have reliable go-around data presented. A direct comparison of the MIL-F-9490D requirements cannot be made with the C-5A.

Discussion

An automatic go-around mode was considered for the C-5A and therefore we feel qualified to comment on this requirement. It is felt that this requirement is valid for present and future aircraft where automatic go-around is required.

Recomendations

Retain the requirement as stated.

Requirement

3.1.2.9.3.2 Lateral-heading AFCS go-around performance standards. The lateral-heading AFCS shall maintain the aircraft 4σ position within the lateral boundaries of the obstacle clearance planes during wind conditions as specified in 3.1.2.9. This capability shall be maintained in the event of the most critical engine failure just prior to or during automatic go-around. If normal procedure is to disengage the go-around mode after inadvertent loss of one engine, under the wind conditions cited a pilot of normal skill shall be able to recover airplane heading such that intersection with the obstacle clearance planes will occur no more than 1 in 10^6 events during recovery.

Comparison

The C-5A does not contain a lateral-heading automatic go-around mode. An automatic go-around mode was originally conceived for the C-5A where the lateral axis was placed in a wings level mode. The primary function of go-around is to achieve the maximum climb gradient in the shortest time. Therefore, a wings level mode for the lateral axes was felt to be the best mode to achieve this goal. The C-5A now contains a manual go-around mode with the capability of using the lateral axes, using CWS, to follow the commands displayed on the flight instruments. The C-5A cannot be directly compared with the MIL-F-9490D requirements.

Discussion

This requirement is valid for present and future aircraft with a change. The first sentence should be changed to include reference to the FAA Advisory Circular 120-29 which is implied. It should be noted that the performance requirement of the last sentence is completely dependent on pilot reaction and performance and is not an operational performance requirement on the AFCS. It does affect the system design of the automatic go-around mode in the area of failure announcement and affect of failures or disengagement of the mode on the aircraft flight path. No change is suggested in this area.

Recommendations

Revise the requirement as follows:

3.1.2.9.3.2 Lateral-heading AFCS go-around performance standards. The lateral-heading AFCS shall maintain the aircraft 4σ position within the lateral boundaries of the obstacle clearance planes defined in FAA Advisory Circular 120-29 during wind conditions as specified in 3.1.2.9. This capability shall be maintained in the event of the most critical engine failure just prior to or during automatic go-around. If normal procedure is to disengage the go-around mode after inadvertent loss of one engine, under the wind conditions cited a pilot of normal skill shall be able to recover airplane heading such that intersection with the obstacle clearance planes will occur no more than 1 in 10^6 events during recovery.

Requirement

3.1.2.9.3.3 Minimum Go-Around Altitude. A minimum altitude for engaging automatic go-around shall be established such that the probability of incurring structural damage to the landing gear, wing tips, or control surface is extremely remote. The minimum altitude shall include normal performance under the wind conditions specified in 3.1.2.9 and the probability of inadvertent loss of an engine at any time within 12 seconds preceding mode engagement.

Comparison

A go-around can be initiated at any time on the C-5A. It was found by analysis and flight testing that no structural damage to the aircraft would occur when the go-around maneuver was initiated at any altitude, even while on the runway.

The C-5A meets the intent of this requirement insofar as structural damage is concerned.

Discussion

This requirement is valid for present and future aircraft with the understanding that it assumes that all aircraft will require a minimum altitude for engaging the go-around mode. Both the C-5A and C-141 flight testing has shown that minimum altitude for these aircraft is the runway altitude.

Recommendation

Retain the requirement as stated.

Requirement

3.1.2.10 All Weather Landing System. The following all weather landing system requirements pertain to the latter stages of the approach; i.e., that portion of the approach below the decision height or the alert height, as defined in 6.6. All weather landing system shall comply with the following landing accuracies:

a. Longitudinal dispersion of the main landing gear touchdown point shall not exceed 1,500 feet with a 2-sigma probability, with a mean touchdown point beyond the projected glideslope intersection with the runway. The 1,500 foot dispersion need not be symmetrically located about the nominal touchdown point. The aircraft sink rate at touchdown shall not exceed the structural limit of the landing gear except as an extremely remote (6.6) occurrence.

b. The lateral dispersion of the aircraft centerline at the main landing gear at touchdown shall not exceed 27 feet on either side of the runway centerline with a 2-sigma probability. The roll out guidance system (normally used during ICAO Category IIIb or IIIc visibility conditions) shall cause the aircraft to track parallel to or convergent with the centerline of the runway.

c. The systems shall meet these requirements considering reasonable combinations of head winds to 25 knots, tail winds to 10 knots, and crosswinds to 15 knots, according to the probability of encountering these winds and their associated turbulence as specified in 3.1.3.7.3, along with expected variations in aircraft configurations as specified in 3.1.2.10.1, and expected variations in ground facility performance as specified in 3.1.2.10.2.

Comparison

The C-5A contains an automatic landing mode which will automatically land the aircraft and control the aircraft along the runway until loss of rudder effectiveness occurs. The mode was evaluated inflight with the following results:

	Mean Value-Ft.	2 Sigma Deviation-Ft.
Longitudinal Distance from glideslope runway intersection	181	802
Lateral Displacement		
from beam center	0.97	26.34
fron runway centerline	3.82	28.48
Sink Rate at touchdown	2.46	2.06

The aircraft sink rate at no time exceeded the structural limits. The C-5A meets the intent of the MIL-F-9490D requirement except for the lateral dispension requirement not to exceed 27 feet from the runway centerline with a 2 sigma probability.

Discussion

Paragraph 6.6 of MIL-F-9490D contains the following definition for All Weather Landing System:

All weather landing system. An all weather landing system includes specifically all the elements of airborne equipment and more generally includes the ground-based equipment necessary for completion of the all weather landing. All weather landings comprise the operations and procedures required to conduct approaches and landings during Category II and III visibility conditions defined by the International Civil Aviation Organization.

This definition states that an AWLS includes all aircraft equipment, ground based equipment, operations and procedures over some of which the contractor has no authority or control. Since this specification is intended to cover the design installation and test of flight controls systems by establishing general performance, design, development and quality assurance requirements for the flight control systems, the requirement for an All Weather Landing System as defined is believed to be beyond the scope of this specification. The majority of the performance requirements stated in the requirements however are pertinent to an automatic landing mode. It is recognized that the procurring activity has the need to exercise its prerogatives for ground and flight procedures and equipment and for weather minimums for which the aircraft should be cleared. The contractor must satisfy the requirements insofar as he is able within the limitations imposed by requirements and equipment over which he has no control. The contractor should therefore be responsible for installing equipment to meet specific performance requirements which are measurable and for which he has control.

Requirement 3.1.2.10b implies that rollout guidance should be designed to accommodate Category IIIb and IIIc visibility conditions. This requirement could require sophisticated ground equipment to be installed at the landing area. The type of ground guidance used would dictate the equipment to be installed in the aircraft. It is felt that this is not feasible since each government organization, aircraft manufacturer, equipment manufacturers and related organizations would have different approaches on proper ground guidance to achieve Category IIIb and IIIc control. In addition, it is believed that there are no commercial or military airfields that have ground equipment that is capable of guiding an aircraft under the stated weather minima. This requirement should require equipment installed which could be used in meeting the FAA Category III a Landing Weather Minima. Advisory Circular AC-120-28A gives the

requirements to obtain approval for this type of system. Any further requirements beyond Category IIIa should be contained in the RFP with an explanation of the ground equipment to be used.

Requirement 3.1.2.10c states the requirements for winds to be met during automatic landing. This paragraph refers to requirement 3.1.3.7.3 for probability of encounter. This paragraph does not clearly define this probability as discussed in the evaluation of requirement 3.1.2.9. The wind shear requirements are not given. It is felt that the wind requirement given in FAA AC-20-57A applies here.

This requirement should be revised to include the FAA requirements contained in AC 20-57A and to be compatible with AC 120-28A. These requirements are capable of being demonstrated and have been accepted by industry. Malfunction and annunciation requirements should also be added.

<u>Recommendation</u>

Revise the requirement as follows:

3.1.2.10 <u>Automatic Landing System</u>. The following automatic landing system requirements pertain to the latter stages of the approach; i.e., that portion of the approach below the decision height or the alert height, as defined in 6.6. Automatic landing system shall be designed to be compatible to operations in ICAO Category IIIa weather minimums and comply with the following landing accuracies and requirement:

a. Longitudinal dispersion about the nominal point of the main landing gear touchdown shall not exceed 1,500 feet with a 2-sigma probability, with a mean touchdown point beyond the projected glideslope intersection with the runway. The 1,500 foot dispersion need not be symmetrically located about the nominal touchdown point. The aircraft sink rate at touchdown shall not exceed the structural limit of the landing gear except as an extremely remote (6.6) occurrence.

b. The lateral dispersion of the aircraft centerline at the main landing gear at touchdown shall not exceed 27 feet on either side of the runway centerline with a 2-sigma probability. The roll out guidance system shall cause the aircraft to track parallel to or converge with the centerline of the runway.

c. The systems shall meet the dispersion requirements considering reasonable combinations of head winds to 25 knots, tail winds to 10 knots, and crosswinds to 15 knots, moderate turbulence, wind shear of 8 knots per 100 feet from 200 feet to touchdown, along with expected variations in aircraft configuration as specified in 3.1.2.10.1 and expected variations in ground facility performance as specified in 3.1.2.10.2.

d. Automatic landing system malfunction should not cause significant displacement of the aircraft from its approach path, including altitude loss or cause any action of the flight control system that is not readily apparent to the pilot, either by control movement or advisory display. Upon system disconnection, the automatic landing system shall not cause any out-of-trim condition not easily controlled by the pilot.

e. Means should be provided to inform the pilot continuously of the mode of operation of the automatic landing system. Indication of system malfunction should be conspicuous and unmistakable. Positive indication should be provided that the flare has (or alternatively has not) been initiated at the minimum normal flare engage heights.

f. The automatic landing system design shall meet the criteria for approval of Category IIIa landing weather minima defined in AC 120-28A.

Requirement

3.1.2.10.1 All weather landing performance standards - variations or aircraft and airborne equipment configurations. All weather landing performance requirements shall be met while including the effects on performance of the following aircraft and airborne equipment variations expected to occur in normal service.

a. Landing weight and center of gravity variations.

b. Landing flap setting variations.

c. Aircraft approach speed variations.

d. Glide slope and localizer airborne receiver centering errors.

e. AFCS all weather landing system sensor, computer, and servoactuator tolerances.

f. Performance tolerances of automatic control systems operating concurrently with the AFCS all weather landing system; e.g., stability augmentation systems, load alleviation systems.

Comparison

The automatic landing on the C-5A was considered a mode of the AFCS. The AFCS was designed to operate under the following conditions.

a. All-weather conditions.

b. With 3 or 4 engines operating and during engine failure.

c. Within the air vehicle gross weight range from 342,000 pounds minimum to 769,000 pounds maximum.

d. The AFCS shall be mechanized such that any single failure will not cause significant upsetting moment or flight path deviation of the air vehicle. Subsystem shutdown or disconnect is permissible.

The automatic throttle system was required to be operational during the automatic landing mode to meet the airplane performance requirements. The AFCS was designed for normal weight and center of gravity variations and landing flap setting variations. The effects of tolerances of all systems used to perform the automatic landing mode were included in the performance analysis.

The C-5A automatic landing mode meets the intent of MIL-F-9490D requirements.

Discussion

This requirement is valid for present and future aircraft except for the title, "All weather landing system". This should be changed to, "Automatic landing system". See the evaluation on requirement 3.1.2.10.

Recommendation

Revise the requirement as follows:

3.1.2.10.1 Automatic landing performance standards - variations or aircraft and airborne equipment configurations. Automatic landing performance requirements shall be met while including the effects on performance of the following aircraft and airborne equipment variations expected to occur in normal service.

a. Landing weight and center of gravity variations.

b. Landing flap setting variations.

c. Aircraft approach speed variations.

d. Glide slope and localizer airborne receiver centering errors.

e. AFCS automatic landing system sensor, computer, and servo-actuator tolerances.

f. Performance tolerances of automatic control systems operating concurrently with the AFCS automatic landing system; e.g., stability augmentation systems, load alleviation systems.

Requirement

3.1.2.10.2 Performance standards - ground based equipment variations. Proof of compliance with performance requirements for all weather landing systems shall include the effects of expected variation in type and quality of the ground based equipment. ILS beam structure, associated tolerances and alignment errors, monitoring, touchdown zone lighting, terrain clearances, and controlled or critical taxi zones shall be considered to meet the requirements for Categories II or III operations as defined by ICAO Annex 10.

Comparison

The performance requirements of the C-5A automatic landing mode included the glide slope beam variations and tolerances. The performance requirements for lateral touchdown were given in displacement from the beam center. It is felt that the C-5A meets the intent of this requirement in regards to ILS beam tolerances.

Discussion

This requirement includes areas that should not be included in a flight control system specification, such as; touchdown zone lighting, and taxi zones. Only flight control requirements that the aircraft manufacturer is responsible for should be included in this specification to insure that compliance with requirements can be domonstrated. This same subject is discussed in the evaluation of requirement 3.1.2.10.

This requirement should include the expected variation of the ILS beam that should be considered during design and evaluation. FAA Advisory Circular AC 20-57A gives beam performance values that have been accepted by industry. These should be included in this requirement by reference.

Recommendation

Revise this requirement as follows:

> 3.1.2.10.2 Performance standards - ground based equipment variations. Proof of compliance with performance requirements for automatic landing systems shall include the effects of expected variation in type and quality of the ground based equipment as defined in FAA Advisory Circular AC 20-57A.

Requirement

3.1.2.11 Flight Load Fatigue Alleviation. A fatigue alleviation control system may be used where it is advantageous to the weapon system. The fatigue alleviation system shall comply with applicable requirements of MIL-A-8866 in addition to the requirements of this specification.

Comparison

The C-5A fleet has been retrofitted with a non-critical active control system called the Active Lift Distribution Control System (ALDCS) which functions as both gust and maneuver load alleviation system to extend the service life of the C-5A inner wing structure. Maneuver load alleviation is attained by the direct effect of aileron inputs which shift the spanwise center of pressure inboard thus reducing the incremental bending moments for a given incremental load factor. Gust load alleviation is attained by reduced short period response of the airframe to continuous turbulence as a result of the inboard elevator pitch damping effect and by the direct lift effect of aileron inputs which reduce the wing first bending responses in turbulence.

Service life improvement is attained through reduced axial tension stresses on the wing lower surface which result from maneuvering and gust load sources. The axial stress reductions are accompanied by increased shear stresses due to the load combinations generated by the aileron inputs (negative bending-positive torsion) thus complicating the analytical prediction of life improvement.

Preliminary estimates of the effectiveness of the ALDCS to provide extended fatigue life have been made using newly developed fatigue analysis methods. These data indicate a fatigue damage rate reduction of approximately 20% or a like extension factor of 1.25 for the life limiting inner wing lower surface structure.

A review of the requirements of MIL-A-8866A revealed no paragraphs directly related to fatigue alleviation control system nor any paragraphs which must be adhered to in order to produce an effective, safe active control system function.

An example of the effect of ALDCS on the first wing bending mode is shown in Figure 1 (3.1.2.11).

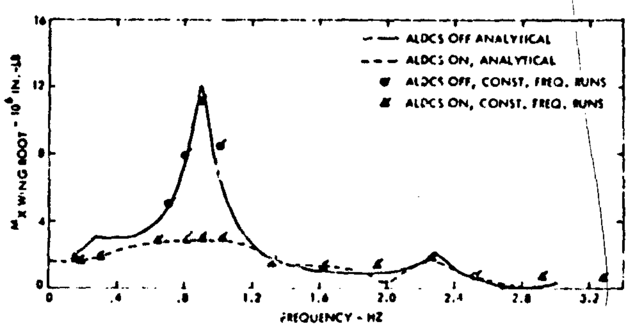


FIGURE 1(3.1.2.11). WING BENDING COMPARISON FOR C-5A AILERON FREQUENCY SWEEPS

Discussion

The fact that paragraphs could not be found relative to fatigue alleviation control systems in MIL-A-8866 leaves the designer wondering which requirements should be addressed. Future revisions to this document may provide specific criteria.

Recommendation

Retain the requirement as stated.

Requirement

3.1.2.12 Ride Smoothing. With the ride smoothing AFCS and other FCS in Operational State I, the following short term and applicable long term vertical or lateral axis ride discomfort index levels shall not be exceeded at any crew station during flight in the turbulence level specified in Table II.

TABLE II
RIDE DISCOMFORT INDEX LIMITS

Ride Discomfort Index, D_i		Flight Phase Duration (Exposure Time)	Probability of Exceeding RMS Turbulence Intensity
Long Term Requirement	0.10	Over 3 Hours	0.20
	0.13	From 1.5 to 3 Hours	0.20
	0.20	From 0.5 to 1.5 Hours	0.20
Short Term Requirement	0.28	Less than 0.5 Hour	0.01

The requirements apply separately to each of the vertical and lateral axes. For the lateral axis requirement only lateral gusts apply and for vertical acceleration only vertical gusts apply. Effects of attitude hold or other pertinent AFCS modes shall be included where used. This requirement normally applies only where a ride smoothing AFCS is specified by the procuring activity. However, where ride smoothing is not specified and other AFCS modes degrade ride quality, the resulting ride shall not degrade to below the levels specified.

3.1.2.12.1 Ride Discomfort Index. Ride discomfort index is defined as:

$$D_i = \left[\int_{0.1}^{f_t} W(f)^2 \; T_{cs}(f)^2 \; \phi_u(f) df \right]^{1/2}$$

D_i = Ride Discomfort Index, (vertical or lateral)

$W(f)$ = Acceleration weighting function (vertical or lateral) 1/g

$T_{cs}(f)$ = Transmissibility, at crew station, g/ft/sec

$\phi_u(f)$ = Von Karman gust power spectral density of intensity specified in 3.1.2.12 and form specified in MIL-F-8785

f = Frequency, Hz

f_t = Truncation frequency (frequency beyond which aeroelastic responses are no longer significant in turbulence) (1)

Acceleration weighting functions are defined for vertical and lateral acceleration by Figure 1. Probability of exceedance versus turbulence intensity is specified in 3.1.3.7.

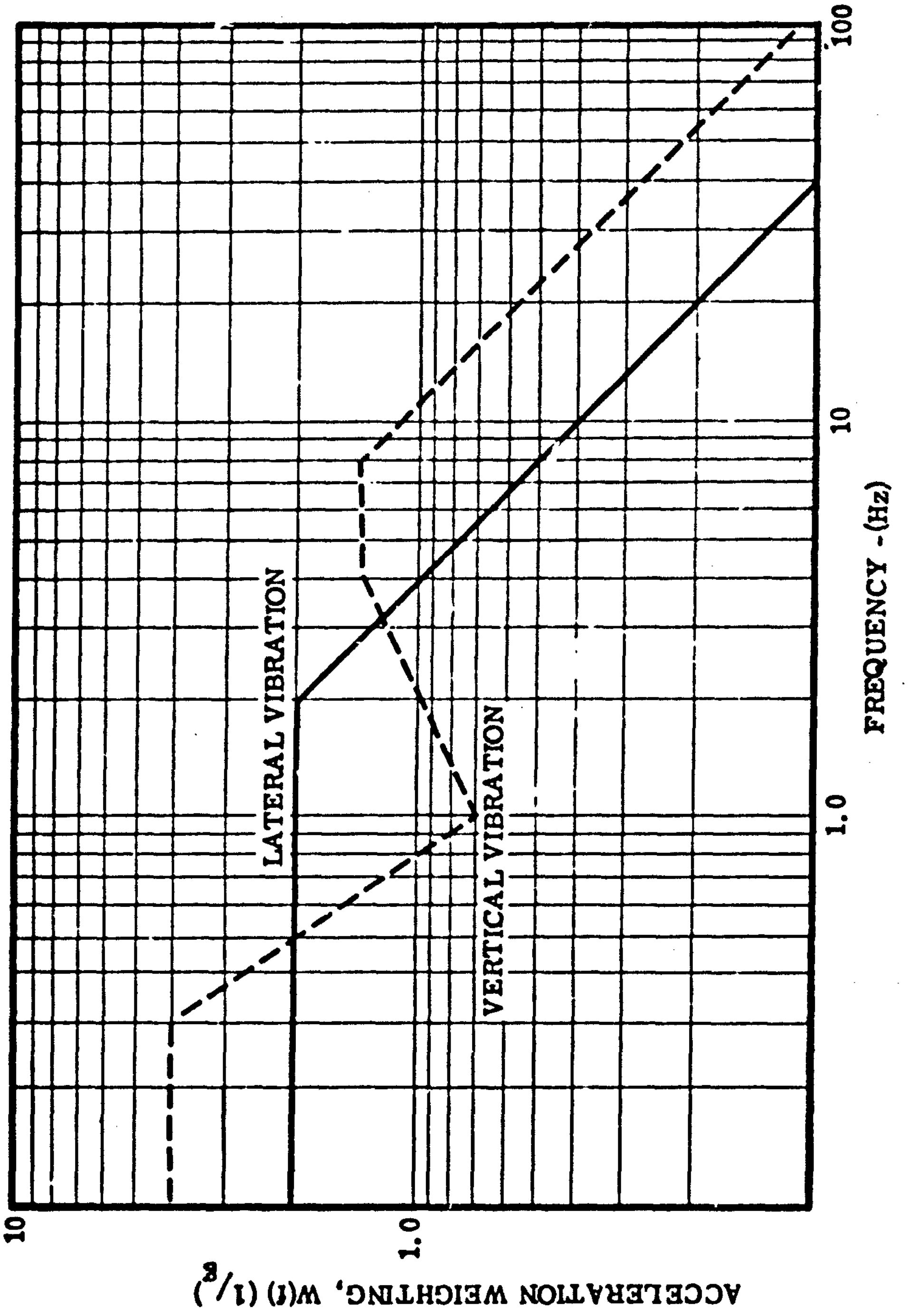


Figure 1. Acceleration weighting functions

Comparison

Lockheed-Georgia generated a digital computer program in 1970 to determine the ride discomfort index as defined in Paragraph 3.1.2.12 for a given airplane and flight condition. Several flight conditions were examined for the C-5A airplane and results showed that in some low altitude, high speed cases, the ride discomfort index (RDI) did not meet the requirements specified in this paragraph. The addition, by retrofit, of an Active Lift Distribution System (ALDCS) has improved C-5A ride qualities significantly. The digital computer analytical program has not yet been modified to include the evaluation of ALDCS effects on the RDI.

Discussion

Lockheed-Georgia has investigated ride quality criteria independently and has generated several digital computer programs for calculating the ride discomfort index, based on the same formulation as that shown in this requirement. This is believed to be the best currently available criterion for ride quality measurement.

It is noted that the index is usually computer per unit RMS gust and then multiplied by the RMS gust value to obtain the RDI number.

It is felt that the long term and short term requirements for ride quality, while valid for transport airplanes, may be overly stringent for fighter/bomber airplanes in certain AFCS modes such as automatic terrain following, where safety of flight and vulnerability to anti-aircraft weapons may take priority over ride quality requirements. If ride smoothing is required by the procuring activity, the function will be designed as an integral part of the basic AFCS and will function during all AFCS modes. The requirements are valid for those aircraft for which the procuring activity may specify ride smoothing. However, when ride smoothing is not specified, the requirement to analyze and verify that the AFCS modes meet the specified RDI levels is too stringent and would result in unnecessary acquisition costs.

Recommendation

It is recommended that the last sentence of 3.1.2.12 "Ride Smoothing" be deleted.

Requirement

3.1.2.13 Active Flutter Suppression. Not Applicable.

Requirement

3.1.2.14 Gust and Maneuver Load Alleviation. An active gust and maneuver load alleviation control system may be used where it is advantageous to the weapon system. The active load alleviation control system shall conform to the applicable requirements of MIL-A-8861 in addition to the requirements of this specification.

Comparison

The initial work on load reduction systems for the C-5A at Lockheed-Georgia began in 1967 and has progressed through several system variations which culminated in the development, production, and installation of the active lift distribution control system. In 1967, Lockheed participated in the aircraft load alleviation and mode stabilization (LAMS) program being conducted by Boeing and Honeywell. Results of the LAMS C-5A system analysis and synthesis showed that an automatic flight control system could reduce structural fatigue damage rates during turbulent flight without significant penalties to basic aircraft stability and handling qualities.

In mid-1969, a program was initiated by Lockheed-Georgia to design, develop, and test a maneuver load reduction system. Rather than a fatigue load reduction system such as the LAMS, the program objective was a system which reduced maximum wing upbending loads, a "strength design" load reduction. The system used symmetric aileron deflections as a function of load factor to shift the spanwise airload distribution. It was named the Maneuver Lift Distribution Control System (MLDCS). The MLDCS also used inboard elevator deflections as a function of symmetric aileron commands to compensate aileron pitching moments.

During the MLDCS development, a requirement was generated to obtain a simplified MLDCS for early fleet incorporation of a load reduction system. The objectives of this system were to reduce wing root bending moments, improve service life by reducing 1g mean bending moments, minimize aircraft performance penalties through maximum use of existing hardware and minimum use of new components. The resulting system, known as the Passive LDCS (PLDCS) used aileron uprig deflection of either 6° ot 12° with the amount of uprig being a function of aircraft configuration and flight condition. The C-5 fleet has been using the PLDCS since November, 1971.

In late 1972, the C-5A Independent Structural Review Team (IRT) included the development of an active LDCS in the list of options available to the Air Force as a means of extending the service life of the C-5A primary wing structure. Air Force and Lockheed review of the IRT options resulted in a joint decision to proceed with an ALDCS development program in mid-1973.

The primary objective of the ALDCS was to provide a system which utilized existing aircraft control surfaces to reduce wing bending moments which result from gust and maneuver load sources.

ALDCS System Design Goals

The following ALDCS system design goals were considered in the design of the computer, sensors, and interfacing with existing aircraft hardware and systems:

The system shall be designed to operate on a "full time basis" within the C-5A speed/altitude operational envelope.

The system shall be designed to "fail safe" concepts.

No single failure of the ALDCS will affect the normal operations of the pitch and yaw/lateral SAS.

The system shall be designed to interface with existing systems and shall use existing sensors and hardware where possible.

The system shall be designed to interface with existing systems and shall use existing sensors and hardware where possible.

The system shall be designed to operate about the present passive LDCS trim positions of the ailerons.

ALDCS Structural Design Goals

The structural goals of the ALDCS specified for gust and maneuver load sources were expressed in terms of load changes at the wing root. Torsion moment increases at other spanwise locations were permitted to exceed the percentage increases specified for the wing root; however, the objective was to minimize the outer wing torsion changes while meeting the criteria specified for the wing root. There were no quantitative requirements for load reductions for discrete gust; however, the ALDCS was required to not increase discrete gust loads.

Continuous turbulence shall result in RMS bending moments at the wing root (W.S. 120) not exceeding 70% of the free airplane values as shown by analyses using the Von Karman gust spectra with a variable scale of turbulence. The above load reductions were required to be realized for a turbulence level of 5 ft/sec RMS. The free airplane did not include the effect of SAS and autopilot.

The RMS gust torsion at the wing root could not exceed the free aircraft values by more than 5%.

The maneuver incremental root bending moments could not exceed 70% of the free aircraft values.

ALDCS Stability and Control Criteria

The ALDCS/Airframe combination was required to meet the following stability margin criteria:

The stability margins shall be such as to preclude:

a. Adverse structural mode coupling
b. Significant degradation in present handling qualities
c. Significant degradation of existing flutter margins
d. Adverse coupling with existing flight control systems
e. Limit cycle tendencies

The stability margin goals are:

a. Ground Test All Modes - 6 db min gain margin and 45 degree phase margin
b. Flight Modes - 6 db min; 10 db goal gain margin and 45 degree min phase margin
c. Flight Modes - beyond control mode natural frequencies - 60 db per decade attenuation (roll-off) and infinite phase margin

The ALDCS shall not produce any significant changes in the existing stability, control and handling qualities.

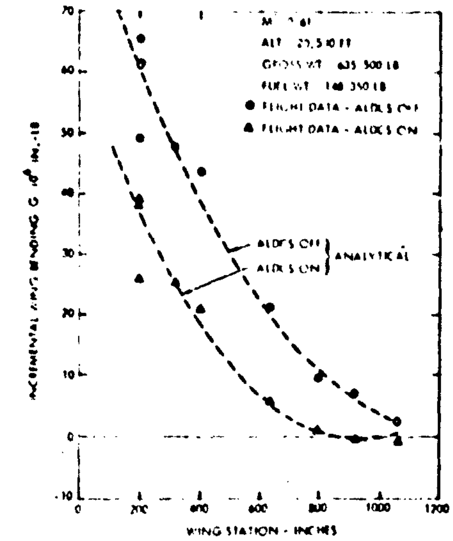
Figure 1 (3.1.2.14) shows the results of flight test using the ALDCS to reduce gust and maneuver loads.

Discussion

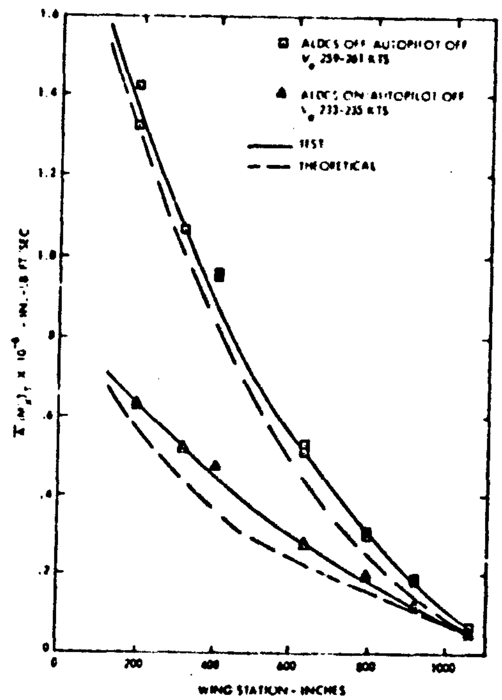
The requirements are non-specific in the area of performance as is appropriate to a system of this nature. The requirement will be applicable to any new large transport and should be retained.

Recommendation

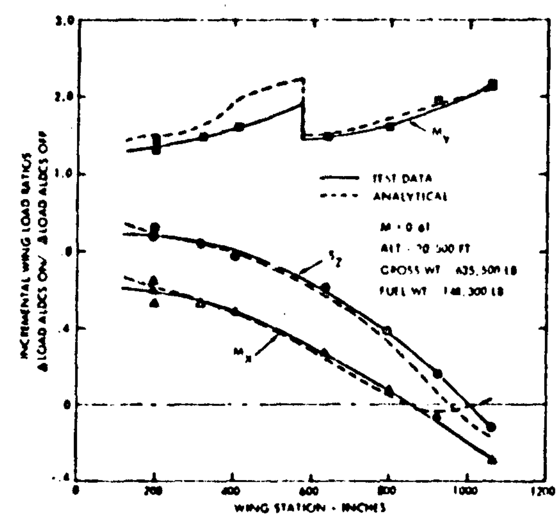
Retain the requirement as stated.



Comparison of Flight Test and Analytical Incremental Bending per G



Normalized RMS Response



Effects of ALDCS on Incremental Wing Loads for Symmetrical Maneuver

FIGURE 1 (3.1.2.14) COMPARISON OF ALDCS FLIGHT TEST AND ANALYTICAL RESULTS

Requirement

3.1.2.15 Automatic Terrain Following. Performance requirements shall be as specified by the procuring activity.

Comparison

The C-5A terrain following system is designed to provide the C-5A aircraft with terrain following capability at airspeeds up to 350 KCAS and a minimum ground clearance of 1,000 feet. This system consists of a multimode radar system, a terrain following computer, and either the pitch axis autopilot or the flight director. The radar supplies forward terrain information to the computer which determines the threatening peaks and generates a flight path angle to maintain the clearance altitude. This computer then sends an error signal to either the autopilot or flight director.

The pilot has the option of selecting ground clearance and one of four ride "hardnesses" which dictate the maximum climb and dive angles. Ride 1 limits the maximum flight path angle to $\pm 7.5°$; Ride 2 limits it to $\pm 5.0°$; and Ride 3 and Ride Soft limit the maximum flight path angle to $\pm 2.5°$. If the threatening peak's range is less than 2.5 miles, the maximum flight path angle can increase by approximately 50%. The rate of change of the flight path angle is limited to 0.9°/second to keep the incremental normal acceleration from exceeding about 0.3 g's.

The system includes an altimeter control command which switches it to replace the terrain following command if the aircraft deviation from its assigned clearance altitude generates a larger error than the terrain following signal or if the radar return is lost. This feature prevents the aircraft from flying dangerously below the assigned altitude.

Discussion

The requirement that the performance of the system be specified by the procuring activity is appropriate for it is only recently that the uniformity has been sought in terrain following criteria. Activities of this nature should be followed to determine their applicability to the specification.

Recommendation

Accept this paragraph "as is."

Requirement

3.1.2.16 Control Stick (or Wheel) Steering. The pilot shall retain full capability to maneuver the airplane within the applicable control force and maneuver limits of MIL-F-8785 or MIL-F-83300. Automatic disengagement of the AFCS with reversion to manual control is permitted in meeting this requirement.

Comparison

The C-5A AFCS has both rate and coupled control wheel steering for both the pitch and roll axes. Rate control wheel steering provides the pilot the capability to maneuver the airplane and establish a new attitude hold reference without disengaging and re-engaging the autopilot. In the coupled CWS mode the pilot inputs are only additions to the autopilot computations. Rate CWS is automatically armed whenever the attitude hold mode is established and the pilot is notified by illumination of the CWS ON lamp on the AFCS control panel. The mode is automatically engaged when either pilot applies a force greater than 2.3 pounds on the control wheel in the nose up or down direction and turning to the left or right.

In the pitch axis, the force level detector circuit on pilot's force sensor triggers the following events when the force level exceeds 2.3 pounds:

a. Pitch attitude and flap inputs to the pitch axis computations are interrupted.
b. The autopilot elevator command is rapidly re-synchronized to prevent transients.
c. Switches close to directly couple the force sensor outputs into the signal chain.

Thereupon the pilot directly controls the amount of elevator commanded and consequently the rate of pitch response by varying the forces applied to the control wheel. Upon removal of the force for one second, the autopilot automatically reverts to the attitude hold mode and synchronizes to the new existing attitude. The roll axis is similarly mechanized.

When engaged in horizontal NAV modes, altitude capture, VERNAV, terrain following, glideslope, or radar approach, coupled control wheel steering can be engaged to allow the pilot to assist the autopilot. Once the primary mode is established, the pilot may arm the CWS function by selecting PITCH CWS (in the case of the pitch axis) on the AFCS control panel. The mode is automatically engaged when forces greater than 2.3 pounds are applied to the wheel. The integrator is braked to prevent its functioning so long as the force is applied. The force signals are directly added into the signal chain causing the aircraft to deviate from the commanded path. Upon release of the force, the autopilot maneuvers the aircraft back to the commanded path of the primary mode.

The pilot must manually disengage the AFCS by one of three ways in order to revert to manual control and have the capability to maneuver the airplane within the applicable control force and maneuver limits of MIL-F-8785. Therefore, the C-5A does not meet this requirement as stated.

Discussion

The Control Wheel Steering modes of an autopilot system are usually individually tailored to suit the aircraft's mission; therefore, designs differ widely from aircraft to aircraft. The decision not to detail design this mode in the flight control specification is commended, but this should be made clear in this requirement.

This requirement as written limits the CWS design to either a disconnect type of system or one which has extra circuitry added to achieve automatic disengagement for reversion to manual control. Since the basic function of the CWS mode is to allow the pilot to change his references for automatic operation, similar to using the roll and pitch mode, then manual disengagement of the autopilot should be considered for meeting the basic requirement of retaining full capability to maneuver the aircraft.

Recommendation

Revise the requirement as follows:

> 3.1.2.16 Control Stick (or Wheel) Steering. The type of control stick (or wheel) steering used shall be provided to the extent specified by the procuring activity. If the mode is provided, the pilot shall retain full capability to maneuver the airplane within the applicable control force and maneuver limits of MIL-F-8785 or MIL-F-83300. Manual or automatic disengagement of the AFCS with reversion to manual control is permitted in meeting this requirement.

Additional Data (For "Users' Guide")

The Control Wheel Steering modes of an autopilot system are usually individually tailored to suit the aircraft's mission; therefore, designs differ widely from aircraft to aircraft. There are fourteen major basic CWS configurations. These configurations are listed in ARINC Report No. 417, dated April 9, 1971, and are given in Figure No. 1 (3.1.5.1.1). The design should select the type or types of CWS that would best meet the aircraft and mission requirements.

Requirement

3.1.3 General FCS Design. Flight control systems shall be as simple, direct, and foolproof as possible, consistent with overall system requirements.

Comparison

The C-5A flight control systems were designed to be as simple and direct as the space available and system performance and redundancy requirements would allow. Cable joints were staggered to prevent wrong connections. Non-standard detail parts and subassemblies were made non-interchangeable by specifying different hole patterns and other different dimensions.

Discussion

The requirement is reasonable and the C-5A meets the requirement. The requirement is specific, but difficult to obtain an objective judgement and proof of its actual accomplishment. Certainly in hindsight most flight control systems including the C-5A can be simplified and in the process improved when judged by these criteria as well as most other criteria. This is a worthwhile design goal which, when kept in mind during the design process, should result in improved reliability, maintenance and operation.

Recommendation

Accept as is.

Requirement

3.1.3.1 Redundancy. The contractor shall determine the redundancy approaches and levels required to satisfy the requirements of this specification.

Comparison

The C-5A Controls System (FCS) was designed to the requirements of Specification No. CP40002-6B titled Performance/Design and Qualification Requirements CEI Number MA0001A C-5A Air Vehicle Flight Controls Subsystem.

The C-5A control system design considered optimization of the requirements for reliability, invulnerability, failure immunity and related safety considerations in meeting the functional system requirements. Variations of mechanical systems, electrical systems, and combinations of these two were used in achieving these design goals. Dissimilar parts and systems were used, where deemed practical, to achieve a more viable redundancy. Normally the degree of redundancy required to achieve the flight phase essential and essential controls was exceeded by one level above the minimum to achieve the reliability, safety, and functional considerations. Specific examples of the C-5A design goals were to provide systems which permit continued operation after any single malfunction, provide aircraft controllability after the loss of two hydraulic systems, include redundancy wherever a failure could seriously degrade safety of flight.

Specific items in MIL-F-9490D will be covered under the paragraph in the specification. See paragraph 3.1.9.4 a(2) for a typical example.

Discussion

The C-5A FCS was designed to requirements that are more specific than redundancy requirements of MIL-F-9490D. This is a good requirement when it is considered as a design goal. The interpretation in this, as in any general specification, has to be supplemented by the configuration application as well as the design guideline "checklist" as defined by the "Users' Guide." Meeting this type of requirement cannot be demonstrated except by abstract non-quantitative terms by opinion based on experience.

Recommendation

Retain the specification as stated. It is suggested that it would be beneficial to expand on degrees of FCS criticality as it may relate to the system design redundancy for more specific examples, for future inclusion in the "Users' Guide." This data could be derived from design/trade studies of existing or future proposed configurations.

Requirement

3.1.3.2 Failure Immunity and Safety. Within the permissible flight envelope, no single failure or failure combination, which is not extremely remote, in the FCS or related subsystems shall result in any of the following effects before a pilot or safety device can be expected to take effective corrective action. For this specification, extremely remote is defined as numerically equal to the maximum aircraft loss rate due to relevant FCS material failures specified in 3.1.7.

a. Flutter, divergence, or other aeroelastic instabilities within the permissible flight envelope of the aircraft, or a structural damping coefficient for any critical flutter mode below the fail-safe stability limit of MIL-A-8870.

b. Uncontrollable motions of the aircraft or maneuvers which exceed limit airframe loads.

c. Inability to land the aircraft safely.

d. Any asymmetric, unsynchronized, unusual operation or lack of operation of flight controls that results in worse than FCS Operational State III.

e. Exceedance of the permissible flight envelope or inability to return to the service flight envelope.

Comparison

The C-5A was shown to be free from aeroelastic instabilities with flight controls in any powered or unpowered condition throughout the flight envelope.

As shown in the validations of Paragraphs 1.2.2 and 1.2.3, the results of multiple failures and even the complete loss of significant portions of a control function for any axis results only in increased pilot workload. Also the pilot actions to detect and counteract these failures are within the normal pilot skills and strength ranges and the override actions required are in the normal sense.

Because of the multiplicity of C-5A control surfaces and control input load paths there is no single failure or probable combination of failures in the FCS which would result in the inability of the pilot to land the aircraft safely. The C-5A exceeded the contracted requirements for failure immunity and flight safety after failures in the FCS.

One of the most significant failures from the standpoint of airframe loads which can be encountered in the C-5A is a hardover pitch axis autopilot and the subsequent recovery therefrom. Flight evaluations of these pitch axis autopilot failures were conducted at the most critical aircraft center of gravity and loading conditions for assessing maximum aircraft positive and

negative g's encountered during the recovery when initial pilot corrective actions were delayed by three second malfunction recognition times. These evaluations showed pilot's recovery of the aircraft resulted in structural loads not more than 80 percent of design limit load factors.

Discussion

The requirements stated are reasonable and achievable.

Recommendation

Retain the requirement as stated.

Requirement

3.1.3.2.1 Automatic Terrain Following Failure Immunity. The terrain following system shall detect any potentially critical failure, not shown to be extremely remote, in the command generation scheme, sensors (including radar and radar altimeter), or terrain following AFCS and provide warning to the pilot. Any failure resulting in loss of the automatic terrain following function or unsafe flying condition shall provide safe exit (automatic fly-up) from the low altitude, high speed environment. Take-over or injection of commands by the pilot while the system is operating shall permit a smooth and positive transition without adverse transients. AFCS function accuracy (heading and roll attitude hold) shall be maintained to the degree specified in 3.1.2.

Comparison

On the C-5A aircraft, terrain following (TF) is provided through the flight director and the autopilot. The terrain following logic and circuits are contained in the multimode radar subsystem. The monitoring system within the multimode radar circuitry provides a fly-up signal or command in the event of a failure or malfunction, thus providing the "safe exit." Appropriate warning lights are illuminated in front of the pilot and copilot. Through the use of CCWS, the pilots are capable of making inputs while the system is operating. In the event of failure, the pilot is signaled to fly-up if in the Flight Director mode. A constant climb is initiated automatically when in the AFCS mode. The autopilot (A/P) or flight director remains coupled with the TF until the mode is disengaged by the pilot. If the problem within the TF system should correct itself prior to pilot disengagement, normal TF mode operation would ensue. In the event of an autopilot malfunction while in the TF mode, disengagement will be automatic with illumination of the A/P PITCH OFF annunciator light and AUTO caution lights. Since the TF mode employs only the pitch channel circuitry, the basic roll axis modes (heading hold, roll attitude hold, or heading select) are available and generally used to complement the operation. The C-5A meets the intent of this requirement.

Discussion

Normally the terrain following function is contained in the radar system and is an independent design from the AFCS, but supplies signals to the AFCS flight director system for manual terrain following and to the autopilot pitch axis for automatic terrain following. This requirement pertains to the requirements between the AFCS and the terrain following system due to a failure in the terrain following mode. The C-5A terrain following system sends a fly-up command to the flight director and to the autopilot in the event of a failure in the TF or if an unsafe flying condition occurs. Some other radar systems that are available only send a fly-up signal in the event of an unsafe flight condition and a warning signal is sent in the event of a failure. In this latter case, the AFCS must generate a fly-up signal within its equipment. The method of producing a fly-up command (within radar or AFCS) must be left up to the individual designer.

A change to this requirement is needed to make it clear that this requirement is on the interface between the AFCS and the terrain following system and does not constitute a requirement to supply terrain following computation or monitoring within the AFCS.

Recommendation

Revise the requirement as follows:

> "3.1.3.2.1 Automatic Terrain Following Failure Immunity. Safe exit (manual and/or automatic fly-up commands) from the low altitude, high speed environment shall be provided for any failure resulting in the loss of the terrain following function or unsafe flying condition. Take-over or injection of commands by the pilot while the system is operating shall permit a smooth and positive transition without adverse transients. AFCS function accuracy (heading and roll attitude hold) shall be maintained to the performance requirements specified in 3.1.2. Failure annunciation of the TF mode shall be provided to the pilot."

Additional Data

The sentences indicated by the left vertical sideline should be added to the background information and "Users' Guide."

Requirement

3.1.3.3 System Operation and Interface. Wherever a noncritical control or any other aircraft subsystem is interfaced with essential or flight phase essential flight control channels, separation and isolation shall be provided to make the probability of propagated or common mode failures extremely remote.

Comparison

In the C-5A aircraft systems which are especially critical for flight control, or which would jeopardize safety of flight if malfunctioning, are provided with built-in limiting devices, emergency disconnects, alternate or redundant systems, and other safety measures as required to insured safe operation of the system.

Discussion

The requirement is applicable to transport aircraft which have "essential and flight phase essential flight control channels" as a qualitative safety of flight inducement. It (the probability) is not a requirement which can be measured and judged as to whether or not an aircraft is in compliance but a guideline for design. Due to the nature of the requirement the lack of stringency is appropriate. This requirement should be valid for design of future transport aircraft without changes.

Recommendation

Accept "as is."

Requirement

3.1.3.3.1 Warmup. After power is applied to the FCS, the warmup time required to meet this specification shall not be more than 90 seconds for MIL-F-8785 Class IV aircraft and not more than 3 minutes for other types of aircraft.

Comparison

The C-5A requirement for warmup is that it not be more than 3 minutes since it is not a MIL-F-8785 Class IV aircraft. From the CEI Specification CP40002-6B the C-5A complies with this requirement.

Discussion

This requirement is applicable to the C-5A and is valid for future transport aircraft. The requirement is complete, practically demonstrable, and has appropriate stringency.

Recommendation

Accept "as is."

Requirement

3.1.3.3.2 Disengagement. Provisions shall be made for positive inflight disengagement of flight phase essential and noncritical electrical controls under all load conditions. No out-of-trim condition shall exist at disengagement which cannot be easily controlled by the pilot. The pilot shall be informed of automatic disengagement. Disengagement circuitry shall be designed such that a failure of the circuitry itself does not prevent automatic or manual disengagement.

Comparison

On the C-5A disengagement is positive under any and all load conditions. Provisions have been made for fail-safe in-flight disengagement and reengagement of the AFCS. An automatic disengagement is signaled to the pilots through appropriate annunciator panels. The systems are designed such that an automatic disengagement or power failure leaves the affected system in its safest condition. A failure in the engage/disengage circuitry causes automatic disengagement.

Discussion

The C-5A at least partially complies with the above requirement. Total compliance may be difficult to ascertain due to the vagueness of the sentence referring to out-of-trim conditions being "easily controlled." This same sentence is worded so that the case of being in an "out-of-trim condition" prior to disengagement is not considered. The requirement is applicable to transport aircraft.

Recommendation

Retain the requirement as stated.

Additional Data

Lockheed has interpreted the second sentence as follows. "Following disengagement, no out-of-trim condition shall result which cannot be controlled by the pilot without exceeding the control forces as defined in Paragraph 3.5.6.2 of MIL-F-8785B."

Requirement

3.1.3.3.3 Mode Compatibility. Mode compatibility logic shall provide flexibility of FCS operation and ease of mode selection. The mode selection logic shall:

a. Make correct mode selection by the crew highly probable.

b. Prevent the engagement of incompatible modes that could create an immediate undesirable situation or hazard.

c. Disconnect appropriate previously engaged modes upon selection of higher priority modes.

d. Provide arming of appropriate modes while certain modes a e engaged.

e. Provide for the engagement of a more basic FCS mode in the event of a failure of a higher priority mode.

Comparison

The C-5A aircraft is equipped with interlocks and mode compatibility logic designed to provide safe and efficient FCS operation. Correct mode selection is encouraged as legend lights are illuminated only as modes become available. The interlock logic prevents the simultaneous engagement of incompatible modes. Engagement of basic or preliminary modes arms advanced modes providing the pilots with choices for further selection. As the higher priority modes are engaged the lower priority ones are automatically overridden. However, when the high priority modes are disengaged, the lower priority modes are available.

Discussion

The C-5A complies with the requirements of this paragraph. The paragraph is clearly applicable to transport aircraft. Compliance with the requirements is easily discernible.

The stringency is appropriate and no changes are seen as necessary to the paragraph.

Recommendation

Accept "as is."

Requirement

3.1.3.3.4 Failure Transients. Aircraft motions following sudden flight control system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. Time delays between the failure and initiation of pilot corrective action shall be as established by MIL-F-8785. Transients due to failures resulting in FCS Operational States I or II within a redundant FCS shall not exceed ±0.5g incremental normal or lateral acceleration at the center-of-gravity of ±10 deg/sec roll rate. Transients due to failures within the FCS resulting in FCS Operational State III shall not exceed 75 percent of limit load factor or 1.5g's from the initial value, whichever is less, at the most severe flight condition.

Comparison

The C-5A flight control systems were designed so that the aircraft limit load factor was not exceeded due to a malfunction. In the case of redundant systems (such as the augmentation systems, the autopilot autoland mode and the mechanical flight controls) the effect of malfunctions had negligible effect on the aircraft load factor or flight path.

The autopilot malfunction tests verified that using reasonable response times the transient produced did not exceed 88% of the aircraft limit load factor. The response times used were identical to those defined in FAA Advisory Circular 25.1329-1A. Hardovers in the pitch axis produced incremental flight load factors of 0.55g down and 0.67g up. In the lateral axis the maximum roll angle reached during the flaps up autopilot malfunction was 25 degrees and increased to approximately 37 degrees during recovery. This bank angle was well below the C-5A limit of 47 degrees and is also within the ±10 deg/sec roll rate limit of this requirement. During flaps down testing the maximum bank angle reached, due to the malfunction and recovery, was 35 degrees. This was below the C-5A limit of 45 degrees and also meets the ±10 deg/sec roll rate limit during the transient of this requirement.

The C-5A meets the intent of this requirement.

Discussion

The requirement that the time delays between failure recognition and initiation of pilot corrective action shall be as established by MIL-F-8785 leaves the requirement open. Paragraph 3.4.9 of MIL-F-8785 does not give the time delays, but only defines what should be considered when determining the time delays. Since the significant time delays in question are pilot reaction times after recognition of the failure, this time should be addressed in this MIL-F-9490D requirement. The pilot's reaction time can be approached differently by different contractors. Pilot reaction times vary; therefore, industry-accepted FAA pilot reaction times can be used to eliminate confusion and to assure safety at least on transport type aircraft. The FAA has defined this time delay in Advisory Circular 25.1329-1A. These FAA time delays are

widely used in the aircraft industry to evaluate the effects on aircraft from malfunction transients and recovery therefrom.

It is recommended that the pilot reaction time contained in AC 25.1329-1A be included in this requirement.

This requirement also addresses redundant systems and failures in systems which render the aircraft in operational State III. The case of a failure in a nonredundant system which leaves the aircraft in operational State I or II after the failure is not covered in this requirement. Since the transient effects of any system failure on the aircraft is independent of of the operational state of the aircraft after the failure, this requirement should only address the transient failure effects of nonredundant and redundant systems as to their effect on the safety of the aircraft. The Operational state of the aircraft after the failure dictates the original design requirement for redundancy.

The requirement to "not exceed 75 percent of limit load factor or 1.5g's from the initial value, whichever is less" is restrictive. Since this specification is a general specification to cover all aircraft, this requirement does not allow the designer the necessary leeway to design to mission, safety and performance requirements. If the effects of transients and recovery therefrom are restricted to not exceeding the limit load factor and shall not cause a dangerous condition then the design is safe. Any further restriction should be contained in the RFP and/or the individual flight control specification.

MIL-F-8785 covers the transient effects of failures in an augmentation system. Since this specification and MIL-F-8785 will probably be required to be met, it is recommended that in the case of augmentation failures, the requirement of MIL-F-8785 be met.

Recommendation

Revise the requirement as follows:

> "3.1.3.3.4 Failure Transients. Aircraft motion following sudden flight control system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. For evaluation of this requirement during climb, cruise, and descent flight regimes, corrective action should not be initiated until three seconds after the pilot has become aware, either through the behavior of the aircraft or failure warning system, that a malfunction has occurred. During maneuvering, approach and landing flight regimes, corrective action should not be initiated until one second after the pilot recognizes that a malfunction has occurred. The malfunction and the subsequent action and airplane recovery should not create

loads in excess of the aircraft design limit loads. For redundant systems the transient effects of the malfunction and subsequent corrective action shall not exceed $\pm 0.5g$ incremental normal or lateral acceleration at the center-of-gravity or ± 10 deg/sec roll rate. Transients caused by failures in augmentation devices shall meet the failure transients requirements of Specification MIL-F-8785."

Requirement

3.1.3.4 System Arrangement. Systems shall be arranged as required to satisfy the reliability, invulnerability, failure immunity and other general requirements of this specification.

Comparison

The C-5A FCS arrangement evolved from all the system requirements defined in the Contract End Item (CEI) specifications of CP40002. There were many requirements and considerations relevant to the FCS arrangement. Refer to the validation discussions for the following referenced paragraphs for the "in depth" objectives. The related paragraphs under 3.1.3 pertain to the general FCS design that specifies "the system shall be simple, direct and foolproof as possible consistent with the overall system requirements." Included were considerations for redundancy, failure immunity, safety, system operation and interface. Other requirements were 3.2.3.1.2 "Separation, Protection and Clearance," 3.2.3.1.1 "Routing," 3.2.3.1.3 "Fouling Prevention," 3.1.9 "Invulnerability," 3.1.6 "Reliability," and 3.1.7 "Quantitative Flight Safety."

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.1.3.5 Trim Controls. Each of the principal control axes shall have trim controls. Wherever worse than Operational State III would result from a power-operated trim control failure that is not extremely remote, the pilot shall be given override capability for the failed control. For series trim control, no worse than Operational State III shall result from a trim control becoming inoperative in any position, except for extremely remote failures. Engagement of the AFCS shall automatically initiate any needed pitch trim. Aircraft subject to short alerts shall have the capability incorporated to return all trim to the takeoff position automatically. Any automatically controlled trim shall incorporate positive means to avoid potentially hazardous adverse trim near stall. In multicrew aircraft with electrical trim systems, interlocks in the circuitry shall prevent simultaneous commands by two aircrew members from causing any operation in opposing directions at the same time.

Comparison

The C-5A is provided with trim in the three principal axes to reduce pilot control forces during prolonged flight conditions.

These trim controls were designed to meet the requirements of CP40002-6B, Performance/Design and Product Confirmation Requirements for the C-5A Air Vehicle Flight Control Subsystem. Requirements were based on MIL-F-9490C and other requirements defined as being necessary to provide safe, reliable and maintainable systems meeting operational requirements in the most effective manner. This specification defined normal and emergency trim system general requirements such as irreversibility under loading and vibratory conditions, trim authority and authority limit adjustments, trim authority with respect to primary control authority, trim rate limits and trim position indication requirements. More detailed definitions of requirements were provided for the longitudinal trim system in areas such as manual and automatic and normal and emergency input command provisions, authority limits, requirements for independent power sources, electrical circuit interlocks to prevent attempts to trim in both directions simultaneously and position indication. The following paragraphs contain discussions of each C-5A trim system configuration and capability.

Aileron Trim - Ref. Figure 1 (3.1.3.5)-- An electro-mechanical trim actuator is located in the input linkage of each aileron control servo assembly such that it is in series with the pilot input system. Trimming is accomplished with the aileron trim knob on the center console. Operation of the aileron trim knob provides an electrical signal to each trim actuator. The aileron trim actuator in turn gives a mechanical input to the aileron control servo assembly, thus providing the desired aileron deflection to maintain wings level flight and allow the pilot's and copilot's control wheel to center. Each aileron trim actuator may be energized separately by operating a switch located to the side of the aileron trim knob. This will provide roll trim in the event one trim actuator is inoperable. The normal aileron trim range

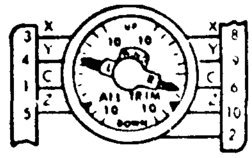
is $\pm 10°$ at a rate of 1/2 degree per second per actuator or a total effective roll trim rate of 1 degree per second. A trim indicator with dual pointers, located in the flight station area, indicates the position of each aileron panel relative to the faired position.

Rudder Trim - Rudder trim is provided by an electromechanical trim actuator attached to the lower rudder input quadrant. The trim actuator for normal operations provides a parallel input to the rudder system. The actuator repositions the natural point of the preloaded centering spring after the rudder pedals have been displaced to a desired trim position. Trim actuator operation is controlled by two rudder trim control switches located on the copilot's side of the center console. The switches are three position (nose left, off, nose right) toggle switches and are spring loaded to the OFF position. Simultaneous operation of the switches is required to provide power and ground signals to the trim actuator. The upper and lower rudder surfaces are trimmed simultaneously as if the input were due to pedal deflection. The trim actuator provides ± 11 degrees trim authority at a rate of one degree per second and trim position is displayed on an indicator located on the center instrument panel.

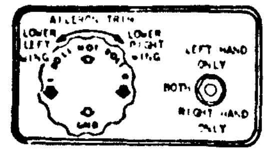
Emergency Rudder Trim Control - Emergency rudder control provides the pilot with ± 10 degrees of upper and lower rudder authority. A YAW AUG MAN TRIM control knob is provided on the Flight Augmentation panel to permit control of the rudders through the Yaw Augmentation (Y/A) subsystem in the event of a jam in the single rudder cable system. A guarded switch to the right of the control knob must be moved from the OFF position to the ON position before the emergency mode becomes operational. Signals are not applied to the Y/A subsystem if the control knob is offset from its neutral position when the guarded switch is thrown to ON. Electrical interlocks are provided which require that the control knob must be returned to neutral position before the signals are switched in. The signals from this control are triple redundant; a failure in one channel will not degrade the system performance.

Horizontal Stabilizer Trim Control System - The pitch trim system includes the horizontal stabilizer actuator and an actuator input system. A high degree of safety and reliability is provided since two signals are required from the input system before the actuator can operate. Figure 2 (3.1.3.5) is a simplified diagram of the entire system.

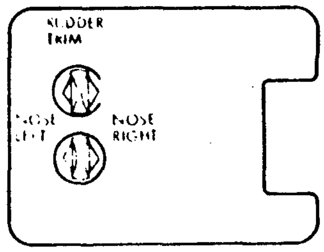
In the design of the pitch trim system, consideration has been given to possible malfunctions and their effect on the controllability of the aircraft. The most dangerous condition is that of a runaway trim actuator. The system has been designed to insure that no single mechanical or electrical malfunction will cause a runaway actuator. However, in the event of a runaway trim actuator, trim disconnect switches are provided as shown in Figure 1 (3.1.3.5) (and discussed in the following text) to immediately disengage power and stop the actuator.



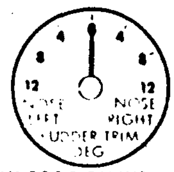
AILERON TRIM INDICATION



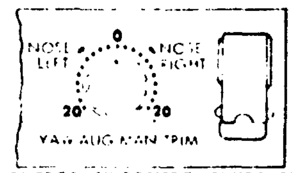
AILERON TRIM CONTROL
CENTER CONSOLE



RUDDER TRIM CONTROL SWITCHES
CENTER CONSOLE (COPILOT'S SIDE)



TRIM POSITION INDICATOR



EMERGENCY CONTROL SWITCHES

C-5A PITCH TRIM SYSTEM

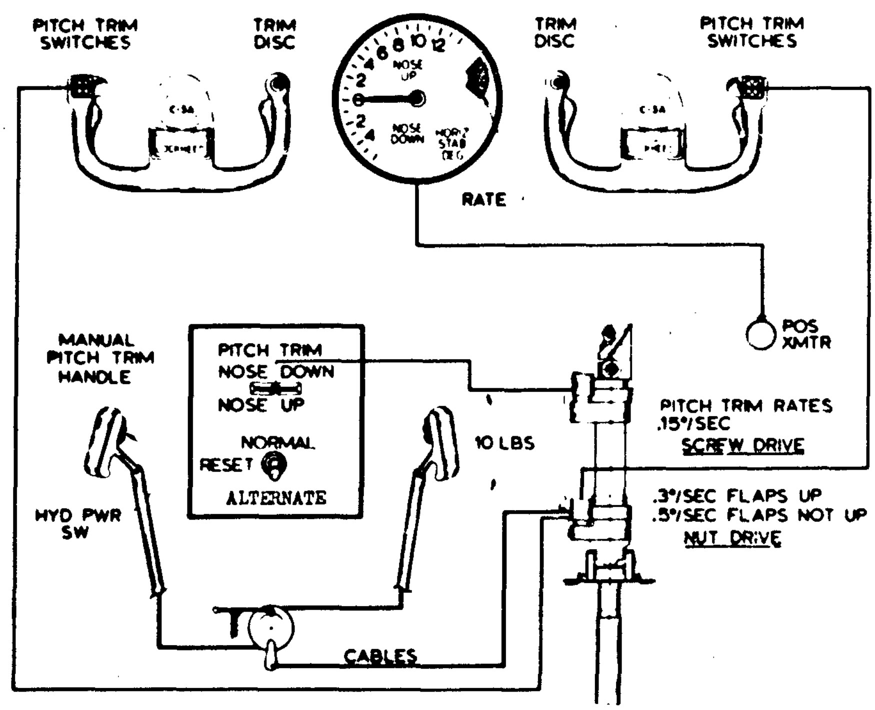


FIGURE 1 (3.1.3.5). C-5A TRIM CONTROLS AND INDICATION

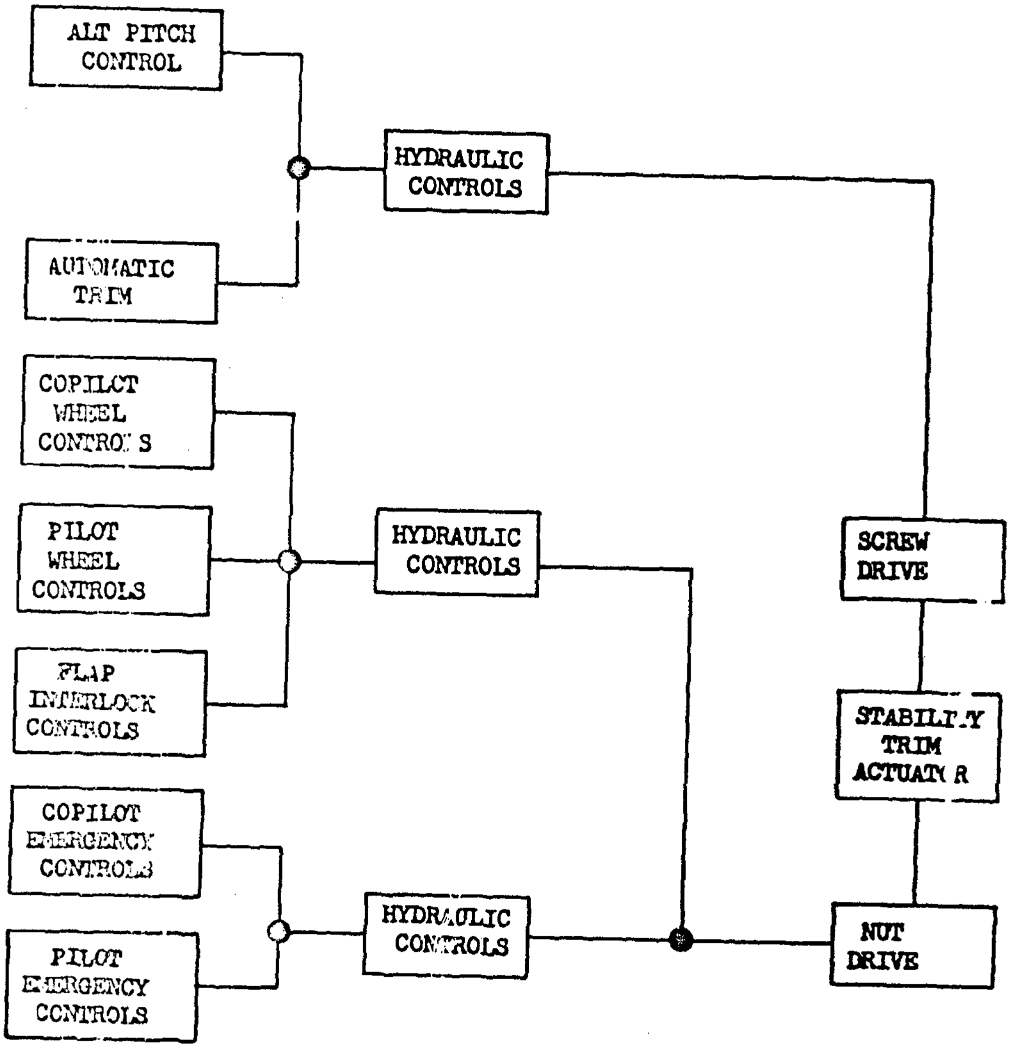


FIGURE NO. 2 (3.1.3.5). STABILIZER TRIM BLOCK DIAGRAM

It is also necessary to insure that no single probable malfunction will result in an inability to trim. The trim system is provided with three separate trim input systems and two separate hydraulic drive systems to assure continued trim capability after any single failure in an input system.

Since there are some single improbable failures in the actuator that could cause an inability to trim and since the actuator is a primary structural member, the main objective in design of the actuator is a positive margin of safety for any loading condition plus a dual load path for the main structural members. The structural integrity of the actuator was further assured through endurance testing of the unit.

Trim about the pitch axis is accomplished by movement of the horizontal stabilizer and is independent of the primary pitch control system (elevators). The pitch trim actuating system consists of the following:

One pitch trim actuator.

Two pilot or copilot operated electrical command systems.

One pilot or copilot operated manual command system.

One autopilot command system.

Four horizontal stabilizer position limit switches.

Two separate hydraulic system inputs.

One horizontal stabilizer position indicator system.

The pitch trim actuator is a dual load path and fail safe actuation concept. Primary design features include minimum end play, irreversibility of the actuator, dual structural load path, dual hydraulic drives (nut and screw) and triple actuation systems. The screw drive unit is commanded by either of two modes:

Alternate mode - The pilot or copilot can operate the pitch trim system in this mode by depressing two switches simultaneously on the center console in the desired direction to send an electrical signal to the screw drive hydraulic manifold shut-off valve and directional control valve.

Autopilot mode - In this mode of operation, the autopilot will send commands to the screw drive hydraulic manifold to operate the trim actuator.

In either of the above modes of operation, the screw will continue to drive until the command signal is stopped or until the horizontal stabilizer operates a limit switch. The limit switch interrupts the command signal and no further trim is possible in that selected direction through the screw drive mechanism.

The nut drive unit is commanded by either of two modes:

Normal mode - Either pilot can operate the pitch trim system by operating dual switches on the outboard grip of either control wheel. This operation completes the circuit to send an electrical signal to the nut drive hydraulic manifold shut-off valve and directional control valve. The nut will continue to drive until the switches are released or until the horizontal stabilizer operates a limit switch.

Manual mode - Either pilot can operate the pitch trim system by moving a lever on either side of the center console and by depressing the trigger switch to send an electrical signal to the nut drive hydraulic manifold shut-off valve which ports fluid to the directional control valve. Moving the lever forward or aft moves the directional control valve to port fluid to the nut drive motor to provide trim in the selected direction. The nut will continue to drive until either the trigger switch is released, the trim handle is brought back to neutral or the actuator reaches the mechanical stops. However, if the aircraft is approaching a stall as evidenced by angle of attack, the Stallimiter circuit will prevent further nose up trim through any mode of pitch trim control. Operating the manual lever switch also sends a 28 VDC signal to the autopilot which disconnects the units while the manual trim system is operating. The manual system can trim the horizontal stabilizer beyond the limit switches until the mechanical stops are reached if the aircraft is airborne. If the aircraft is on the ground the signal to the hydraulic shutoff valve is routed through the 12° stabilizer down and the 1.50° stabilizer up limit switches thereby limiting the stabilizer travel on the ground.

Two horizontal stabilizer limit switches mounted on the vertical stabilizer control the trim limits in the stabilizer leading edge up direction as a function of flap position and the aerial refueling door being open or closed. Two other limit switches control the trim limits in the stabilizer leading edge down direction as a function of flap position only. The limit switches do not affect the manual trim mode.

<u>Autopilot Trim Control System</u> - The autopilot trim mode is operated by autopilot trim commands and uses the same relays as the alternate trim mode. The autopilot trim mode is provided to relieve the autopilot primary pitch circuit (elevators) from having to put in a continuous pitch signal to trim the aircraft.

Pitch Trim Disconnect Circuit - Located on the inboard grip of the pilot and copilot control wheels is a pitch trim disconnect switch which disconnects the 28 VDC power source from all trim control system modes, except the manual mode. These trim modes will remain disconnected until a reset switch, located above the alternate trim mode switch, is operated.

The reset switch has three positions. The center position is neutral. The up position operates the relay connecting the normal trim system. The down position operates the relay connecting the alternate and autopilot systems. Comparison of the C-5A pitch trim system with MIL-F-9490D shows that where it complies with the requirements that the following specific conditions are met:

1. The autopilot trim mode will provide inputs to the pitch trim actuator when the autopilot is "on."

2. The Stallimiter System will interrupt the nose up trim system when the aircraft approaches a stall and avoid adverse trim. There would be no hazardous condition since the C-5A elevator is capable of providing a maneuver load factor of 1.5 against the most adverse aircraft nose down stabilizer trim position at the design dive speed.

3. There is a relay in the trim circuit that disconnects when opposite direction inputs for trim is called for to prevent dual signals calling for opposite direction trim.

4. One failure in the trim system will not reduce trim capability but may change the procedure for trimming so that the pitch trim system is no worse than Operational State II.

Comparison of the aileron trim system with the MIL-F-9490D requirements shows compliance with the intent since after the failure of one actuator there is adequate control available from the actuator in the opposite wing to counteract the resulting rolling moment to maintain control no worse than Operational State III.

Comparison of the rudder trim system with the MIL-F-9490D requirements shows compliance with the intent since after a failure in the normal trim system, the emergency trim control has sufficient authority to provide yaw trim so that the only new pilot action required is a change in his trim procedure. This is no worse than Operational State II.

Discussion

The requirements contained in paragraph 3.1.3.5 are directed at operational capability and are important insofar as they go in relating to the problems associated with trim systems design. However, comparing the C-5A requirements

to these requirements reveals areas not covered in the current MIL-F-9490D requirement which are most important considerations in the design of trim systems (particularly for longitudinal trim of large transport aircraft) such as:

Redundancy of structural load paths

Redundancy of power sources

Irreversibility under loading and vibratory conditions

Lockheed concludes that the stated requirement is valid, is met by the C-5A, can be practically demonstrated, but it is not complete enough to assure the desired end results.

Recommendations

Revise the requirement as follows:

The trim system shall be designed with a normal trim range not in excess of the absolute total requirements for the air vehicle. The primary control system shall have sufficient power to control and land the air vehicle with the trim system positioned for the most adverse trim requirement encountered during the normal cruise flight envelope and mission requirements. Trim authority greater than the primary control system capability, for a given configuration, shall be avoided where mission requirements permit.

Trim actuators shall be irreversible so that loads or vibratory conditions will not alter trim settings and shall maintain a given setting until changed by intentional commands. Electrical trim limit switches may be incorporated to permit adequate adjustment of the normal trim travel.

The trim actuator irreversible mechanism shall be located and designed to minimize free play and maintain rigidity in the control. Where a failure of a power-operated trim control system would result in marginal or unsafe control characteristics, other control capability such as a completely separate emergency trim system, and means to override the failed system, shall be provided to the pilots. In determining an acceptable trim rate to meet the manual flight requirements, the following shall be considered in addition to the requirements stated herein:

a. The trim system rates shall meet the flying qualities requirements of MIL-F-8785(B).

b. The maximum trim rate for Class I, II and III aircraft should be no greater than that which creates a maneuver to give limit airframe load during aircraft recovery from a trim runaway after a three second delay in recognition and response for corrective action.

Trim rates should be kept as low as possible, consistent with a. and b. above. Rates of application shall be such that precision of control is obtained for landing, takeoff, and inflight conditions without creating a hazard. When series trim is used, the system shall be designed to ensure manual control through the pilots' controls in the event that the actuator becomes inoperative in any position.

The longitudinal trim system shall be designed to accept pilot manual commands from the cockpit controls and from automatic flight control subsystems and shall have sufficient motion to meet the trim requirements. The elevator shall be capable of providing a load factor of 1.5 against the most adverse longitudinal trim position at the design dive speed. No single failure shall result in a runaway of any normal or alternate trim systems. The trim input systems shall be designed to accept necessary automatic trim signals and to provide a maximum automatic trim rate which will not produce objectionable transients. The automatic trim arrangement shall be such that a runaway can be counteracted with normal control action, can be arrested by manual trim or disconnected.

Requirement

3.1.3.6 Stability. For FCS using feedback systems, the stability as specified in 3.1.3.6.1 shall be provided. Alternatively, when approved by the procuring activity, the stability defined by the contractor through the sensitivity analyses of 3.1.3.6.2 shall be provided. Where analysis is used to demonstrate compliance with these stability requirements, the effects of major system nonlinearities shall be included.

3.1.3.6.1 Stability Margins. Required gain and phase margins about nominal are specified in Table III for all aerodynamically closed loop FCS. With these gain or phase variations included, no oscillatory instabilities shall exist with amplitudes greater than those allowed for residual oscillations in 3.1.3.8, and any nonoscillatory divergence of the aircraft shall remain within the applicable limits of MIL-F-8785 or MIL-F-83300. AFCS loops shall be stable with these gain or phase variations included for any amplitudes greater than those allowed for residual oscillations in 3.1.3.8. In multiple loop systems, variations shall be made with all gain and phase values in the feedback paths held at nominal values except for the path under investigation. A path is defined to include those elements connecting a sensor to a force or moment producer. For both aerodynamic and nonaerodynamic closed loops, at least 6 db gain margin shall exist at zero airspeed. At the end of system wear tests, at least 4.5 db gain margin shall exist for all loops at zero airspeed. The margins specified by Table III shall be maintained under flight conditions of most adverse center-of-gravity, mass distribution, and external store configuration throughout the operational envelope and during ground operations.

TABLE III
GAIN AND PHASE MARGIN REQUIREMENTS (DB, DEGREES)

Mode Frequency Hz \ Airspeed	Below $V_{O_{MIN}}$	$V_{O_{MIN}}$ To $V_{O_{MAX}}$	At Limit Airspeed (V_L)	At 1.15 V_L
f_M <0.06	GM = 6 DB (No Phase Requirement Below $V_{O_{MIN}}$)	GM = ±4.5 PM = ±30	GM = ±3.0 PM = ±20	GM = 0 PM = 0 (Stable at Nominal Phase and Gain)
0.06 ≤ f_M <First Aero-Elastic Mode		GM = ±6.0 PM = ±45	GM = ±4.5 PM = ±30	
f_M >First Aero-Elastic Mode		GM = ±8.0 PM = ±60	GM = ±6.0 PM = ±45	

where: V_L = Limit Airspeed (MIL-A-8860).

$V_{O_{MIN}}$ = Minimum Operational Airspeed (MIL-F-8785).

$V_{O_{MAX}}$ = Maximum Operational Airspeed (MIL-F-8785).

Mode = A characteristic aeroelastic response of the aircraft as described by an aeroelastic characteristic root of the coupled aircraft/FCS dynamic equation-of-motion.

GM = Gain Margin = The minimum change in loop gain, at nominal phase, which results in an instability beyond that allowed as a residual oscillation.

PM = Phase Margin The minimum change in phase, at nominal loop gain, which results in an instability.

f_M = Mode frequency in Hz (FCS engaged).

Nominal Phase and Gain = The contractor's best estimate or measurement of FCS and aircraft phase and gain characteristics available at the time of requirement verification. (2)

Comparison

C-5A FCS were analyzed using conventional definitions of phase and gain margin, i.e., open loop analysis methods, not the definitions proposed in Table III of Paragraph 3.1.3.6.1. Where more than one crossing of the zero DB or -180 degree lines occurred, the minimum phase and gain margins (most adverse) were used to define the system stability along with information obtained by transient response and root solution analysis methods.

Specifically, for the C-5A ALDCS System the following requirements were established:

A. The stability margins shall be such as to preclude:

1. adverse structural mode coupling
2. significant degradation in present handling qualities
3. significant degradation of existing flutter margins
4. adverse coupling with existing flight control systems
5. limit cycle tendencies

B. The stability margin goals were:

1. Ground test (all modes) 6 db min. gain margin and 45 degree min. phase margin

2. Flight modes: 6 db min., 10 db goal gain margin, 45 degree min. phase margin

3. Flight modes - beyond control mode natural frequencies: 60 db per decade attenuation (roll-off) and infinite phase margin

Discussion

The definitions of phase and gain margin given in Table III are ambiguous in that these parameters are not generally considered to vary with frequency as reported in the User's Guide. It is accepted practice to use most adverse (minimum) margins where more than one crossing of the zero DB or -180 degree lines occur on the open loop frequency responses. Relating margins with each of the natural modes is an unwieldy procedure and the bandwidth divisions are arbitrary and difficult to substantiate.

Recommendation

Revise the requirement as follows:

"3.1.3.6.1 Stability Margins. Required gain and phase margins about nominal are specified in Table III for all aerodynamically closed loop FCS. For both aerodynamic and nonaerodynamic closed loops, at least 6 db gain margin and 45° phase margin shall exist at zero airspeed. With these gain or phase variations included, no oscillatory instabilities shall exist with amplitudes greater than those allowed for residual oscillations in 3.1.3.8, and any nonoscillatory divergence of the aircraft shall remain within the applicable limits of MIL-F-8785 or MIL-F-83300. AFCS loops shall be stable with these gain or phase variations included for any amplitudes greater than those allowed for residual oscillations in 3.1.3.8. In multiple loop systems, variations shall be made with all gain and phase values in the feedback paths held at nominal values except for the path under investigation. A path is defined to include those elements connecting a sensor to a force or moment producer. At the end of system wear tests, at least 4.5 db gain margin shall exist for all loops at zero airspeed. The margins specified by Table III shall be maintained under flight conditions of most adverse center-of-gravity, mass distribution, and external store configuration throughout the operational envelope and during ground operations.

Revise Table III as follows:

TABLE III

GAIN AND PHASE MARGIN REQUIREMENTS

Airspeed	Below $V_{O_{MIN}}$	$V_{O_{MIN}}$ To $V_{O_{MAX}}$	At Limit Airspeed (V_L)	At 1.15 V_L
Gain Margin	±6 DB	±6 DB	±4.5 DB	0 DB
Phase Margin	±45°	±45°	±30°	0°

Where V_L = Limit Airspeed (MIL-A-8860)

$V_{O_{MIN}}$ = Minimum Operational Airspeed (MIL-F-8785)

$V_{O_{MAX}}$ = Maximum Operational Airspeed (MIL-F-8785)

Gain Margin = The minimum change in loop gain, at nominal phase, which results in an instability beyond that allowed as a residual oscillation.

Phase Margin = The minimum change in phase, at nominal loop gain, which results in an instability.

Nominal Phase and Gain = The contractor's best estimate or measurement of FCS and aircraft phase and gain characteristics available at the time of requirement verification.

Requirement

3.1.3.6.2 Sensitivity Analysis. Tolerances on feedback gain and phase shall be established at the system level based on the anticipated range of gain and phase errors which will exist between nominal test values or predictions and in-service operation due to such factors as poorly defined nonlinear and higher order dynamics, anticipated manufacturing tolerances, aging, wear, maintenance and noncritical material failures. Gain and phase margins shall be defined, based on these tolerances, which will assure satisfactory operation in fleet usage. These gain and phase tolerances shall be established based on variations in system characteristics either anticipated or allowed by component or subsystem specification. The contractor shall establish, with the approval of the procuring agency, the range of variation to be considered based on a selected probability of exceedance for each type of variation. The contractor shall select the exceedance probability based on the criticality of the flight control function being provided. The stability requirements established through this sensitivity analysis shall not be less than 50 percent of the magnitude and phase requirements of 3.1.3.6.1.

Comparison

On the C-5A ALDCS system, component manufacturing tolerances and calibration limits were applied to each path from sensor to control surface to obtain information on degradation of stability and performance. The basic stability margins stated in the previous paragraph were required to be maintained with these tolerances and limits applied.

Discussion

This paragraph is acceptable as is, can be demonstrated, and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.1.3.7 Operation in Turbulence. In Operational State I, while flying in the following applicable random and discrete turbulence environment, the FCS shall provide a safe level of operation and maintain mission-accomplishment capability. For essential and flight phase essential controls, at least Operational State III shall be provided in the specified flight-safety turbulence levels. Noncritical controls shall provide at least Operational State II in turbulence up to the intensities specified in 3.1.3.7.1. Noncritical controls operating in turbulence at intensities above the specified turbulence level, shall not degrade flight safety or mission effectiveness below the level that would exist with the control inactive. Either manual or automatic means to inactivate the control for flight in heavy turbulence may be used, when required. The dynamic analysis or other means used to satisfy this requirement shall include the effects of rigid body motion, significant flexible degrees of freedom and the flight control system. Significant nonlinear effects shall be represented by conservative nonlinear or equivalent linear representations.

Comparison

The C-5A aircraft design specification, CP 40002, does not include specific requirements which relate to flight control system operation in turbulence. Although not subjected to tests to verify compliance with the paragraph, it is believed that the C-5A's performance satisfies the requirements.

Discussion

The requirements for essential and flight phase essential controls are appropriately severe. The stringency of the requirement for noncritical controls is less as it should be. This requirement would be very difficult to evaluate in actual flight test. Dynamic analysis would almost have to be the means employed for compliance demonstration. Even this technique would require a rather complex simulation to fully cover the requirement. However, turbulence operation is critical to aircraft safety of flight, and requirements pertaining to it should be included in future transport design.

Recommendation

Accept the paragraph "as is."

Requirement

3.1.3.7.1 Random turbulence. The RMS turbulence intensity to be used for normal flight and for terrain following shall have a cumulative probability of exceedance as specified in table IV.

TABLE IV.
TURBULENCE INTENSITY EXCEEDANCE PROBABILITY

FCS Function Criticality \ Aircraft Class	MIL-F-8785 Class III	MIL-F-8785 Class I, II & IV
Essential	10^{-6}	10^{-5}
Flight Phase Essential	$\frac{1}{T}10^{-6}$	$\frac{1}{T}10^{-5}$
Noncritical	10^{-2}	10^{-2}

where:

T = the longest time spent in essential flight phase segment in any mission/total flight time per mission. (3)

Table V specifies RMS vertical gust amplitude versus altitude for selected exceedance probabilities. The relationship among vertical, lateral and longitudinal RMS intensities and scales as specified in MIL-F-8785 shall be used to establish intensities for lateral and longitudinal gusts. The listed turbulence intensity levels apply at the turbulence penetration airspeed V_G. At the maximum level flight airspeed, V_H these intensity levels are reduced to 38 percent of the specified levels. The mathematical forms of continuous random turbulence to be used in conjunction with the specified intensity levels are as specified in MIL-F-8785 and the airspeeds cited are as specified in MIL-A-8860.

TABLE V
RMS GUST INTENSITIES FOR SELECTED CUMULATIVE EXCEEDANCE PROBABILITIES, FT/SEC TAS

FLIGHT SEGMENT	ALTITUDE (FT – AGL)	PROBABILITY OF EXCEEDANCE						
		2×10^{-1}	10^{-1}	10^{-2}	10^{-3}	10^{-4}	10^{-5}	10^{-6}
TERRAIN FOLLOWING	UP TO 1000 (LATERAL)	4.0	5.1	8.0	10.2	12.1	14.0	23.1
	UP TO 1000 (VERTICAL)	3.5	4.4	7.0	8.9	10.5	12.1	17.5
NORMAL FLIGHT CLIMB CRUISE AND DESCENT	500	3.2	4.2	6.6	8.6	11.8	15.6	18.7
	1,750	2.2	3.6	6.9	9.6	13.0	17.6	21.5
	3,750	1.5	3.3	7.4	10.6	16.0	23.0	28.4
	7,500	0	1.6	6.7	10.1	15.1	23.6	30.2
	15,000	0	0	4.6	8.0	11.6	22.1	30.7
	25,000	0	0	2.7	6.6	9.7	20.0	31.0
	35,000	0	0	0.4	5.0	8.1	16.0	25.2
	45,000	0	0	0	4.2	8.2	15.1	23.1
	55,000	0	0	0	2.7	7.9	12.1	17.5
	65,000	0	0	0	0	4.9	7.9	10.7
	75,000	0	0	0	0	3.2	6.2	8.4
	OVER 80,000	0	0	0	0	2.1	5.1	7.2

3.1.3.7.2 Discrete gusts. Discrete gust amplitudes to be used shall be established using the relationship between random and discrete gust amplitudes in accordance with MIL-F-8785, and the RMS amplitudes specified in 3.1.3.7.1. The 1-cosine discrete gusts in accordance with MIL-F-8785 shall be applied with wavelengths tuned to provide maximum excitation.

Comparison

Since the C-5A aircraft design specification, CP-40002, does not include FCS analysis and design requirements for turbulence operation, the C-5A was not analyzed with respect to this requirement. Gust analyses performed generally have been in compliance with the Dryden formulas for random and discrete gusts as given in MIL-F-8785. The C-5A turbulence analysis, therefore, may or may not have complied with the requirements of these paragraphs when performed.

Discussion

The requirements for a standard and consistent methodology for implementation of random and discrete gust models are valid. Ease of evaluation and comparison is enhanced through consistency in the models. The references to MIL-F-8785 also enhance the uniformity of the turbulence requirements as this puts both military specifications in agreement on a common turbulence model. The paragraph's stringency is justifiable as another effort to maintain uniformity. Compliance with the specific requirements of the paragraphs is not difficult to accomplish. The paragraphs are well stated and clearly applicable to future transport design as they now read.

Recommendation

Accept both paragraphs as they are presently stated.

Requirement

3.1.3.7.3 Wind model for landing and takeoff. The following wind model form shall be used for automatic navigation and all weather landing system design as required by 3.1.2.9 and 3.1.2.10. This model applies for low altitude approach and landing flight phases at conventional airports and shall not be applied at heights greater than 500 feet above mean runway level.

3.1.3.7.3.1 Mean wind. The probability of occurrence of total mean wind and mean crosswind components is shown on figure 2 as a function of wind speed in knots as measured at a reference altitude of 20 feet above mean surface level.

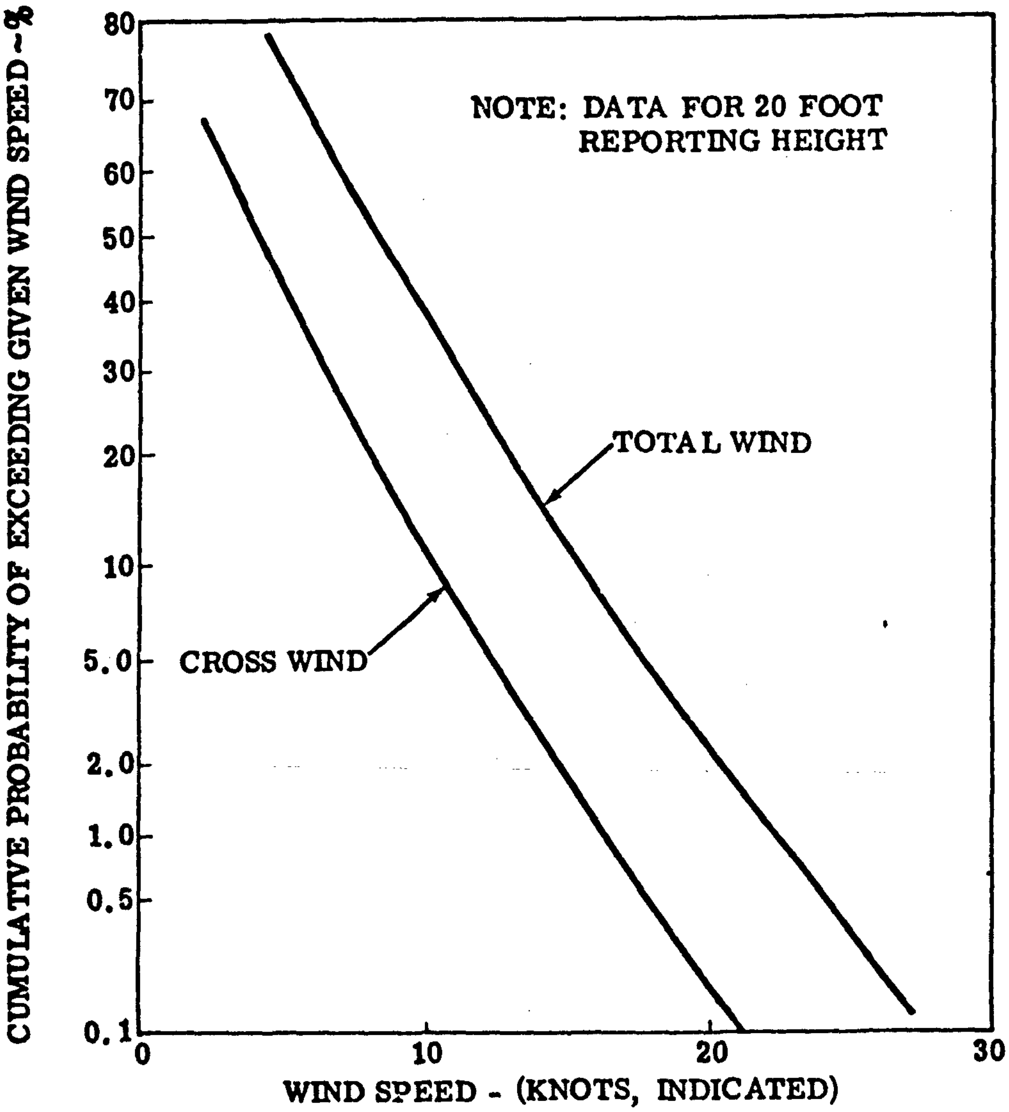


Figure 2. Cumulative probability of report mean wind and crosswind when landing

Comparison

The wind model as described in these paragraphs was not used to verify the requirements of Paragraphs 3.1.2.9 and 3.1.2.10 for the C-5A. The simulation analysis of the C-5A auto-land mode in approach and landing used a variety of gust models. Most were constant cross, head or tail wind components. For random and discrete gusts the criteria for Dryden models as given in MIL-F-8785 were used. Therefore, the C-5A does not comply with these requirements.

Discussion

The requirement is valid because it calls for a standard or uniform wind model to be used in examining compliance with the specifications. Compliance is easily demonstrated. The requirement for use of a consistent model is applicable to future transport aircraft. The first sentence of 3.1.3.7.3.a is not precisely compatible with the title of Figure 2 of the requirement.

Recommendation

Revise the first sentence of the requirement as follows:

> "3.1.3.7.3.1 Mean Wind. The probability of exceeding reported mean wind is shown in Figure 2....

Requirement

3.1.3.7.3.2 Wind Shear. Wind shear shall be included in each simulated approach and landing unless its effect can be accounted for separately. The magnitude of the shear is defined by the expression

$$u = .46\ U \log_{10}(Z) + .4\ U$$

where: u = mean wind at height Z feet in feet/sec (true)

U = mean wind at 20 feet in feet/sec (true)

Z = height above gound (feet) (4)

Comparison

The C-5A aircraft All Weather Landing System (AWLS) was analyzed using simulation techniques. Gust disturbances in the form of wind shears were included in these analyses. However, the method employed to determine magnitudes is not in compliance with the paragraph's expression but is believed to be acceptable to show compliance with mission and safety requirements.

Discussion

An attempt was made to utilize this wind shear equation and it is not clear how this equation and the curve of mean wind exceedance are used together to determine mean wind, U. Some questions that needed to be answered are: how is the formula used; is the mean wind u or U the same as that defined in paragraph 3.1.3.7.3 ; if so why are the units not the same?

This area of probability of occurence and wind shear must be made completely clear to the user of MIL-F-9490D.

Recommendation

Perform an investigation into the proper use of the equations given and revise this requirement to clarify the terms and to define how the equations should be used.

Requirement

3.1.3.7.3.3 Wind model turbulence. The longitudinal wind component (in the direction of the mean wind) and vertical and lateral wind components shall each be represented by a Gaussian process having a spectral density, $\Phi(\Omega)$, of:

$$\Phi(\Omega) = \sigma_i^2 \frac{2L_i}{\pi} \frac{1}{(1 + \Omega^2 L_i^2)}, \left(\frac{FT}{SEC}\right)^2 \Big/ \frac{RAD}{FT}$$

where: σ_i = RMS turbulence level in an axis in feet/sec

L_i = Scale length in an axis, feet

Ω = Spatial frequency in radians/ft. (5)

and the value for σ and L is shown on table VI.

TABLE VI
RMS TURBULENCE LEVEL AND SCALE LENGTH BY AXIS

	Vertical	Lateral	Longitudinal
σ	0.1 U	0.2 U	0.2 U
L	15 Ft for $Z \leq 30$ Ft .5 Z Ft for $30 \leq Z \leq 1000$	600 Ft 1000 Ft	600 Ft

Comparison

During the All Weather Landing System (AWLS) simulation analysis of the C-5A some vertical random gusts were input in the longitudinal cases. The turbulence models employed were generated according to the Dryden formulae as specified in MIL-F-87[illegible] . No attempt was made to comply with MIL-F-9490 turbulence specificati[illegible].

Discussion

As stated, this paragraph gives a clear and concise requirement for wind model turbulence analysis. Compliance can be practically demonstrated. Stringency of the requirement is appropriate as it is intended to bring uniformity to gust models used in simulation analyses of approaches and landings. No change in the requirement is seen as necessary to make it applicable to future transport aircraft.

Recommendation

Accept "as is."

Requirement

3.1.3.8 Residual Oscillations. For normal operation and during steady flight, FCS induced aircraft residual oscillations at all crew and passenger stations shall not exceed 0.04g's vertical or 0.02g's lateral peak to peak acceleration. Residual oscillations in pitch attitude angle shall satisfy the longitudinal maneuvering characteristic requirements of MIL-F-8785. Residual oscillations in roll and yaw attitude at the pilot's station shall not exceed 0.6 degree peak to peak for flight phases requiring precision control of attitude.

Comparison

The C-5A flight control system was designed to meet the following requirements:

- All control surfaces and surfaces such as flaps shall be free from any tendency toward undamped oscillations apparent to the pilot under specified flight conditions.

- Residual oscillations, induced by the AFCS, as measured in the cockpit during steady flight shall not produce normal acceleration in excess of ± 0.02, lateral acceleration in excess of ± 0.01g, pitch attitude amplitudes in excess of ± 0.1 degree, yaw attitude amplitudes in excess of ± 0.1 degree and roll attitude amplitudes in excess of ± 0.15 degrees.

During the C-5A flight testing, no residual oscillation was evidenced by the data or perceptable to the pilot with and without the AFCS engaged.

It is felt that the C-5A meets the intent of this requirement.

Discussion

This requirement is valid for present and future aircraft. It is recommended that the residual oscillation limits in pitch attitude angle be included in this requirement in lieu of referencing MIL-F-8785. Paragraph 3.2.2.1.3 of MIL-F-8785 requires that the pitch angle residual oscillations be no greater than ± 3 mils. This is equivalent to $\pm .17$ degree.

Recommendation

Revise this requirement as follows:

"3.1.3.8 Residual Oscillations. For normal operation and during steady flight, FCS induced aircraft residual oscillations at all crew and passenger stations shall not exceed 0.04g's vertical or 0.02g's lateral peak to peak acceleration. Residual oscillations in roll and yaw attitude shall not exceed 0.6 degree and in pitch attitude shall not exceed 0.17 degrees peak to peak at the pilot's station for flight phases requiring precision control of attitude.

Requirement

3.1.3.9 System Test and Monitoring Provisions. Test and monitoring means shall be incorporated into the essential and flight phase essential FCS as required to meet the following requirements of this specification:

Mission Reliability	3.1.6
Flight Safety	3.1.7 to 3.1.7.1
Fault Isolation	3.1.10.2 to 3.1.10.2.2
Failure Immunity & Safety	3.1.3.2 to 3.1.3.2.1
Survivability	3.1.8 to 3.1.8.1
Invulnerability	3.1.9 to 3.1.9.7

The effect of detected and undetected FCS failures taken with the probability of occurrence of such failures shall comply with the system reliability and safety requirements. This requirement shall include all failures, both active and latent, and failures in all components of the system, including mechanical, electrical and hydraulic components.

Comparison

The C-5A has built-in test circuitry for the following purposes:

1. Functional self-test for ground pre-flight and system maintenance
2. Approach test of comparators during all weather landing phase

The systems to which built-in test procedures are applied include:

- o pitch, roll and yaw augmentation
- o pitch, roll and yaw autopilot
- o pilot assist cable servos (PACS)
- o all weather landing system
- o auto-throttle system
- o Flight Director
- o Stallimiter
- o Go-Around System
- o ALDCS
- o CADC

Discussion

The C-5A built-in test provisions cover some functions which were ultimately demonstrated to be non-critical. These test provisions do, however, facilitate maintenance and enhance mission reliability. The requirement is valid and easily demonstrated.

Recommendation

Accept "as is."

Requirement

3.1.3.9.1 Built-in Test Equipment (BIT). The total maintenance aid testing, including BIT, and in-flight monitoring where used, shall provide an integrated means of fault isolation to the LRU level with a confidence factor of 90 percent or greater. BIT functions shall have multiple provisions to ensure they cannot be engaged in flight. The test equipment shall not have the capability of imposing signals which exceed operating limits on any part of the system or which reduces its endurance capability or fatigue life. Ground test signals shall not be of sufficient magnitude to drive actuators into hard-stop limits.

Comparison

The C-5A flight controls self-test capability includes an interface with the MADARS (Malfunction and Detection Analysis and Recording System) to provide rapid ground and in-flight isolation of malfunctions to the LRU level.

Discussion

It is felt that the fault isolation requirement in this paragraph rightly belongs in Paragraph 3.1.10.2, since this paragraph is referenced in 3.1.3.9. Further, the term "confidence level" is not considered as definitive as "probability of detection" in current usage. It is recommended that the first sentence in this paragraph be deleted and that Paragraph 3.1.10.2 be modified to require a "90% probability for detecting failures" as indicated in the comments on that paragraph.

Recommendations

A. Revise this requirement as follows:

> 3.1.3.9.1 Built-in Test Equipment (BIT). BIT functions intended for ground check out only shall have multiple provisions to ensure they cannot be engaged in flight. The test equipment shall not have the capability of imposing signals which exceed operating limits on any part of the system or which reduces its endurance capability or fatigue life. Ground test signals shall not be of sufficient magnitude to drive actuators into hard-stop limits.

B. Revise requirement 3.1.10.2 per recommendation.

Requirement

3.1.3.9.1.1 Preflight or Pre-Engage BIT. Preflight or pre-engage BIT may be automatic or pilot-initiated and includes any test sequence normally conducted prior to take-off or prior to engagement of a control to provide assurance of subsequent system safety and operability. It should be demonstrated that redundant MFCS electronic channels are operating normally without any safety-critical latent failures prior to take-off. This includes all backup or normally disengaged channels and fault monitoring and failure isolation elements. The preflight tests shall not rely on special ground test equipment for their successful completion. Any test sequence which could disturb the normal activity of the aircraft in a given mode shall be inhibited when that mode is engaged.

Comparison

The C-5A was designed with BIT as one of the main objectives. All the AFCS contain BIT and do not rely on any special ground test equipment. At no time during any of the BIT is the normal activity of the aircraft disturbed. All the computers contain the capability of testing the system on the face of the computer. For a few systems, additional test switches are located in the cockpit so that the BIT can be initiated by the pilot. The C-5A does contain one automatic pre-engage BIT which is engaged during the automatic landing mode. When the auto-land mode is engaged, automatic test of the autopilot circuitry to be used is performed after glideslope engagement. The following chart lists the systems that contain BIT and which have the capability of test initiation in the cockpit.

	Initiation Location	
System	On Computer	In Cockpit
Stallimeter	Yes	Yes
SAS	Yes	Yes
ALDCS	Yes	No
Autopilot	Yes	No
Auto Throttle	Yes	No
Flight Director	Yes	Yes
Go-Around	Yes	No
CADC	Yes	Yes

Discussion

The main points covered in this paragraph are automatic or pilot-initiated tests allowed, special ground equipment requirements, safety and operability assurance, testing of redundant MFCS electronic channels, and disturbance of normal in-flight system operation by test equipment. These requirements are adequately covered by the present phraseology.

Recommendation

Retain the requirement as stated.

Requirement

3.1.3.9.1.2 Maintenance BIT. Where required, BIT shall also be provided as a postflight maintenance aid for the FCS. BIT shall be designed to avoid duplicating test features included as part of the preflight test or monitoring functions.

Comparison

The C-5A has BIT for functional post-flight and maintenance testing for all automatic flight control modes.

Discussion

The wording of this paragraph is considered a little vague in regard to what is required when the procurement document calls for maintenance BIT.

Recommendation

Delete the last sentence and substitute the following:

> "Insofar as practical, the BIT used for pre-flight and pre-engage testing should be utilized for post-flight maintenance purposes. The prospective contractor should specify any additional BIT required for maintenance, along with its impact on cost, weight, maintainability and reliability."

Requirement

3.1.3.9.2 Inflight Monitoring. Continuous monitoring of equipment performance and critical flight conditions shall be active, as a minimum, during essential or flight phase essential modes of operation. False monitoring warnings, including the automatic or normal pilot response thereto, shall not constitute a specific hazard in excess of the system reliability requirements.

Comparison

All C-5A FCS have been designed so that the affect of any single failure is not critical. In each of the hydraulic actuator and/or motors the hydraulic power supply is monitored and if any hydraulic supply is lost the Master Caution light is illuminated. The annunciator lights located between the pilot and copilot will illuminate indicating which function is affected. The overhead panel will indicate what part of the function that has sustained the failure. The rest of the mechanical system is monitored by pilot operation, observation of surface defective indicators and by control and aircraft response. The C-5A AFCS sends signals to the MADAR system for annunciation and recording of failures that occur.

Discussion

This requirement as stated could be interpreted to require automatic monitoring to be added to mechanical flight controls even though such monitoring is not necessary either for maintenance or for performance requirements. In addition, the extent that automatic inflight monitoring is used for malfunction detection and isolation should be determined by the complexity of the systems required to perform the mission requirements. This requirement should specify the need for inflight monitoring, but should not dictate that it must be supplied for all the FCS.

Recommendation

Revise the requirement as follows:

> "3.1.3.9.2 Inflight Monitoring. Continuous inflight monitoring of essential or flight phase essential electronic equipment performance shall be active, as a minimum, during critical flight conditions. Continuous inflight monitoring of essential or flight phase essential pilot operated controls shall be active as a minimum on equipment for which failure cannot be determined by observation or normal operation of the control. False monitoring warnings, including the automatic or normal pilot response thereto, shall not constitute a specific hazard."

Requirement

3.1.4 MFCS Design. The following general requirements apply. References to mechanical or electrical MFCS apply only when the mechanization is used:

a. Augmentation. When used, augmentation systems shall be compatible with all control modes and airframe dynamic considerations. Single failures in a gain scheduling system, not classed as extremely remote, shall not degrade augmentation system performance below Operational State II. Pilot-operated gain changing devices shall only be used as emergency backup equipment. Specific approval shall be obtained from the procuring activity for this feature. Positive mechanical or electrical stops shall be provided in gain schedulers to preclude exceeding limiting gain values.

b. Ratio Changing Mechanisms. Where ratio changing mechanisms are used, monitors and emergency positioning means shall be provided if improper positioning can result in a safety of flight hazard.

c. Control Centering, Breakout Forces and Free Play. The corresponding design requirements of MIL-F-8785 or MIL-F-83300 shall be met. Selected sensitivity and breakout forces shall not lead to overcontrol tendencies.

d. Reversion. If a backup mode is provided for a flight control system, at least FCS Operational State III shall be provided following reversion. While disengaged, interaction of backup mode provisions with the normal mode shall not degrade operation below State I. If a single FCS power system is used in an essential or flight phase essential fully powered system, emergency mechanical reversion or an emergency power source shall be provided. On single-engine aircraft, the emergency power source shall be independent of engine operation. It shall be possible to re-engage the normal power source in flight following operation with manual reversion controls or emergency power. Manual or automatic changeover to or from emergency provisions shall not result in capability worse than FCS Operational State III.

e. Controller Kinematics. Kinematics shall preclude hazardous unintentional inputs (crosstalk) into one or more axes with normal control motions within the limits of ultimate structural load factor, design maneuver and turbulence induced accelerations experienced at the crew station.

f. Feedback to Crew Station Controls. The control device motion and force required to accomplish stability and control augmentation shall not be evident at the crew station controls. Vibratory forces or motion acting upon elements downstream of the controller shall not be evident at the crew station controls. Force and motion feedback to crew station controls shall be considered as not evident if the force magnitude is less than half the lowest breakout force of the applicable control.

Comparison

a. Augmentation. Stability Augmentation Systems (SAS) operational throughout the C-5A maneuvering envelope are employed in the three primary flight control axes. They are triple redundant; i.e., fail operational, fail safe systems. As seen in Table I(1.2.3), no single failure in any axis of the SAS results in SAS performance below Operational State II. The Yaw and Lateral functions of stability augmentation are contained within the same computer but each axis may be operated individually. Packaging within the same box eliminates duplication of rate signal source and the functions performed are interrelated to the extent that performance of either is improved when simultaneous operation is effected. The Yaw functions include yaw damping, dutch roll damping and turn coordination. The Lateral (roll) functions include spiral divergence control and roll damping when the autopilot is engaged. The roll/yaw autopilot requires a functioning Y/A subsystem for engagement. The pitch augmentation subsystem provides short period pitch damping which is required for A/P operation. Appropriate interlocks are provided to prevent autopilot pitch axis engagement without a functioning pitch augmentation subsystem. The system may be engaged during manual flight without noticeable effect on handling qualities.

Discussion

The C-5A SAS does not contain any pilot operated gain changing devices. SAS are sufficiently monitored and limited to insure that gain value limits and airplane design load factors are not exceeded and that a system malfunction or combination of probable malfunctions does not cause a catastrophic failure. Necessary controls and displays are provided to the pilots for ground checkout and in flight operations. The requirement is valid; its stringency is justified; it is met by the C-5A and it is practically demonstrable.

Recommendation

Accept as is.

Comparison

b. Ratio Changing Mechanisms. The Aileron-Spoiler Mixer Assembly is a ratio changing mechanism with functions as follows:

1. Change the relationship between flight spoiler deflection with control wheel deflection as a function of flap position.

2. Up rig the flight spoilers 3° when the flaps extend beyond approach.

3. During a roll maneuver with the flight spoilers up rigged 3°, the mix box will allow the flight spoilers on the up going wing to go down 3° to the faired position with 20° of control wheel rotation. After the flight spoilers have reached the faired position, a dwelling motion will occur to allow additional down aileron commands without affecting the aileron feel forces.

4. Allow the ground spoiler cable system to deploy the flight spoilers without affecting aileron panel position.

5. Allow the use of the ailerons and flight spoilers for roll control without affecting the ground spoiler panel position after ground spoiler deployment.

Discussion

In the failure analysis in Report LG1US42-2-1, Volume V, the worst failure in the mix box results in a system jam in one wing. After shearing pins in the links between the L.H. and R.H. systems, the remaining system will provide adequate roll control (See Roll Control System Figure II-4. If the aileron or roll control spoilers are deflected to a large angle when the jam occurs, it may be desirable to shut off hydraulic pressure to these units thereby increasing the possibility of the surfaces floating to a more favorable position. The pilots will know which wing is jammed after shear out because the pilots control will be operable if right wing controls are inoperable and the copilots control will be operable if the left wing controls are inoperable. The C-5A meets this requirement which is valid and can be practically demonstrated.

Recommendation

Retain the requirement as stated.

Comparison

c. Control Centering, Breakout and Free Play. The following compares the C-5A data with the MIL-F-8785 requirement, Paragraph 3.5.2.1 and 3.5.2.2

	Measured C-5A Lb.	MIL-F-8785 Class III Max. Lb.
Elevator Push	5	7
Pull	6	7
Aileron	6	6
Rudder	10-13	14

The free play of the C-5A control systems is held to a minimum by use of cable tension regulators, minimum number of pin joints and close tolerances pin joints.

Discussion

The breakout forces of the C-5A are within the requirements and include a centering spring force that is greater than system friction to ensure positive system centering. The free play is small and does not result in objectionable surface dead band characteristics or overcontrol tendencies. The C-5A meets this requirement which is reasonable and compliance can be shown easily.

Recommendation

Accept as is.

Comparison

d. Reversion. Each C-5A Control System has two or more hydraulic systems to supply power. If a hydraulic system is lost, the remaining system(s) supply power (See Figure II-2 for hydraulic power distribution). Each primary servo control manifold is equipped with shut-off valves so that each hydraulic system may be shut "off" and "on" without affecting the remaining system(s). The switches are located on the Overhead Panel in the Flight Station.

Discussion

There is no immediate pilot action to be taken when a hydraulic system is lost since all systems operate full time and each hydraulic system provides a part of the total power. The loss of one hydraulic system does not result in capability worse than Operational State II. Pilot selection of by-pass for the lost hydraulic system may be accomplished as a low priority task. The C-5A meets the requirement.

Recommendation

Retain the requirement as stated.

Comparison

e. Controller Kinematics. The C-5A control system routing, system separation and structural rigidity prevent unintentional inputs (crosstalk) between systems. Intentional cross coupling is provided between the

primary control axes through SAS. For instance, elevator inputs are required to offset pitching moments produced by symmetrical deployment of ailerons in response to gust and maneuver load alleviation commands from the Active Lift Distribution Control System (ALDCS).

Discussion

The C-5A meets the requirement.

Recommendation

Accept as is.

Comparison

f. Feedback to Crew Controls. The C-5A Stability Augmentation System is mechanized by using a concentric sleeve type valve. The pilot input opens the valve by moving the valve spool. Control surface movements are proportional to movements of the pilots' controllers. Stability Augmentation inputs move the control valve sleeve. Stability augmenting control surface movements are independent of pilots' controller movements except as constrained by surface stops.

Figure 1 (3.1.4) depicts a control servo hydraulic schematic which is representative of C-5A primary control servos.

Discussion

The force from stability augmentation commands fed back into the pilots' input system is equal to control valve friction. This feedback is a minute fractional part of the system breakout force. The C-5A meets this requirement.

Recommendation

Accept as is.

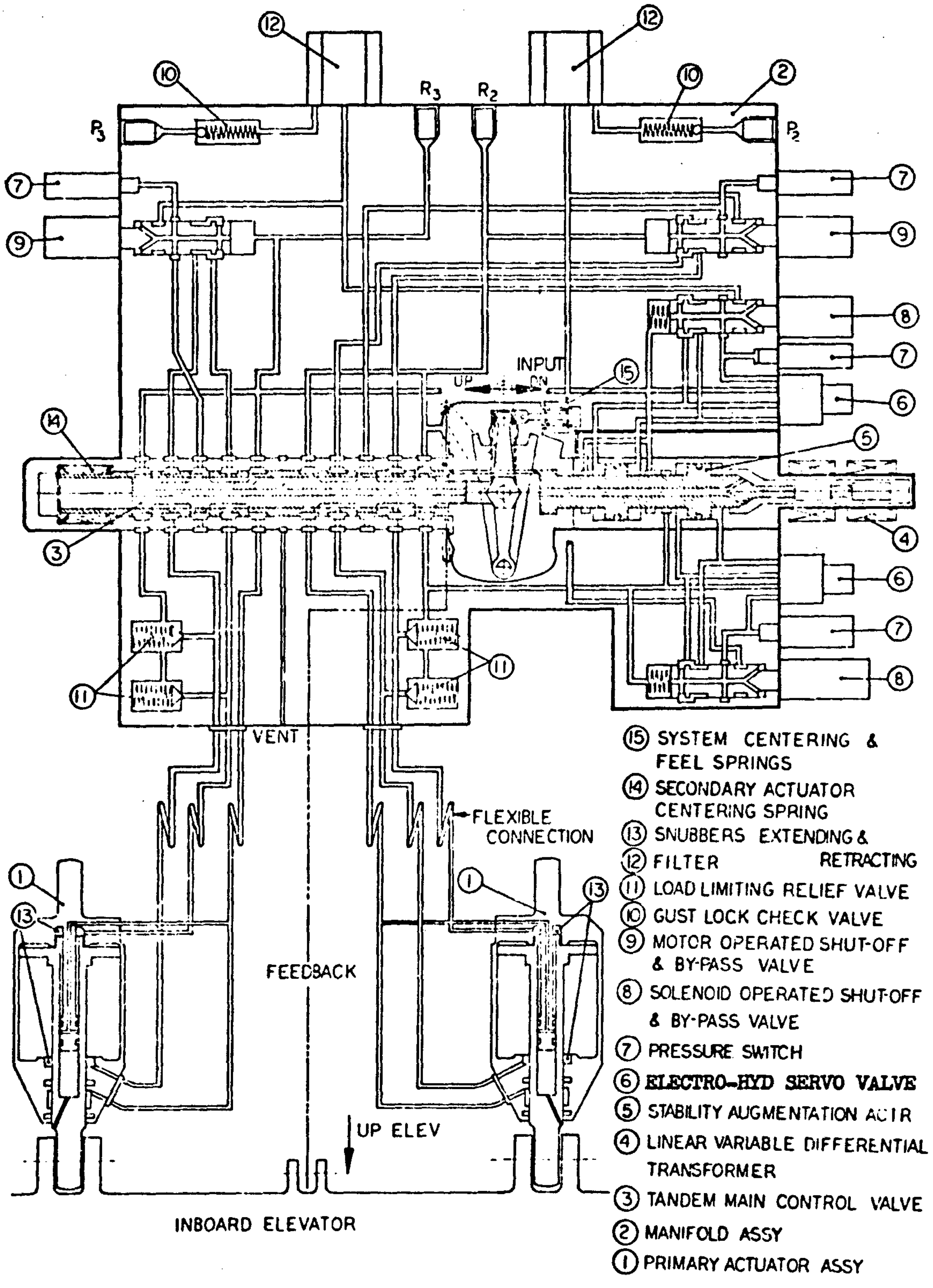


FIGURE 1 (3.1.4). HYDRAULIC SCHEMATIC INBOARD ELEVATOR SERVO

Requirement

3.1.4.1 Mechanical MFCS Design. In the design of mechanical components, the reliability, strength and simplicity of the system shall be paramount considerations. The signal transmission between the pilot's controls and the control surfaces shall be redundant to the extent required to meet reliability, failure immunity, invulnerability and other requirements of this specification.

Comparison

The mechanical elements of the C-5A FCS were designed to achieve the objectives outlined in the validation discussions for Paragraphs 3.1.6 "Mission Accomplishment Reliability," 3.1.11 "Structural Integrity (Load Capability, Strength, Stiffness, and Durability)," and the requirements of 3.1.3 "General FCS Design" to achieve simplicity and operation as foolproof as possible.

The mechanical FCS achieved a level of redundancy, between the pilots controls and the control surface actuation, to meet the requirement for 3.1.3.2 failure immunity and safety - where a single failure or failure combination would not cause FCS or aircraft degradation below the requirements specified therein. The MFCS met the 3.1.6 reliability requirement for preclusion of the probability of mission failure based on FCS failure, including power supplies and also met the 3.1.9 invulnerability requirement which limited the degradation of the FCS due to induced environment, other system failure, crew error or enemy action. The redundancy requirement of 3.1.3.1 provided a MFCS which would permit continued operation after any single malfunction and redundant design wherever a failure could involve safety of flight. Figures No. 1, 2, and 3 (3.2.3.1.1) depict examples of MFCS redundancy and system separation (per 3.2.3.1.2).

Discussion

This is a valid requirement. It has been satisfied by the C-5A FCS design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.1.4.1.1 Reversion - Boosted Systems. In the mechanical reversion mode, at least FCS Operational State III shall be provided. The normal, boosted, control forces shall provide FCS Operational State I. It shall be possible to re-engage boost following operation with mechanical reversion.

Comparison

This requirement is applicable to the C-5A pitch and roll control power assist cable servo (PACS) only since all the control surfaces are powered by multiple system, fully powered hydraulic servo actuators.

Discussion

This is a good requirement which should be applicable to boosted or powered systems where reversion to a mechanical mode is required to meet the specified criteria. The level of mechanical redundancy should be determined by criteria such as failure, redundancy, reliability, survivability, and invulnerability. The C-5A satisfies the requirement.

Recommendation

Retain the requirement as stated.

Requirement

3.1.4.2 Electrical MFCS design. Electrical flight control systems (6.6) shall be designed with special consideration to invulnerability to lightning strikes and to the thermal, EMI and other induced environments of 3.1.9.3

Comparison

The C-5A FCS hardware consists essentially of mechanical input systems controlling control surface power units. The C-5A FCS does contain provisions for non-critical SAS and ALDCS which are commanded electrically. The C-5A FCS does not have an electrical (fly by wire) control system as the primary control mode, therefore this specification is not applicable. However, Lockheed is currently involved in associated internal development programs and is therefore desirous of commenting.

Discussion

An electrical control system provides series inputs to the primary servos for stability augmentation (SAS). Although the electrical input SAS system was not the primary control mode, the system was designed to consider the invulnerability to induced environments. The SAS system is an active/standby system and provides the redundancy and monitoring to protect against failures and hardover. In addition, the primary mechanical input system was always "in the loop" to neutralize any hardover failure transient of the SAS, resulting from a multiple failure. The Lockheed-Georgia Company experience with electrical MFCS (fly by wire) has included the FCS for the "Hummingbird" vertical take-off aircraft and numerous R and D studies and proposals which included "active/standby" and "active/active" systems such as hybrid and redundant-voting. Currently a force summing FCS concept is being tested with prototype hardware for application to future improved state of the art aircraft such as CCV.

Lockheed agrees with the emphasis on designing the electrical MFCS with special consideration to invulnerability to lightning strikes, thermal effects, EMI and other induced environments as defined in requirement paragraph 3.1.9.2. In fact all of the invulnerability requirements under paragraph 3.1.9 should warrant special consideration in designing this type of FCS.

Recommendation

Revise the requirement as follows:

> 3.1.4.2 Electrical MFCS design. Electrical flight control systems (6.6) shall be designed to meet the invulnerability requirements under 3.1.9 with special consideration to invulnerability to lightning strikes and to the thermal, EMI, and other induced environments of 3.1.9.3.

Requirement

3.1.4.2.1 Use of Mechanical Linkages. If a separate artificial feel system is used, or if mechanical linkages are used to connect a signal conversion mechanism with the control surface actuators, friction and freeplay shall not result in FCS operation below State I. Longitudinal and directional controls shall be mass balanced in the fore and aft direction and lateral controls shall be provided inboard to outboard balance, consistent with structural mode and longitudinal force requirements. Any residual vertical imbalance shall be consistent with feel requirements.

Comparison

The C-5A FCS elevator variable feel unit (VFU) provides artificial feel for pitch control. Figure No. 3 (3.2.3.1.1) shows the artificial "Q" feel connected to the co-pilot's input quadrant at the base of the control column. The elevator artificial feel schematic, Figure No. 1 (3.2.4.2), shows the tie in for all its related functions. The mechanical linkage connecting the signal transmission uses close tolerance connections and a preloaded cam mechanism which minimizes free play. The VFU is located as close as practical to the pilots controllers to minimize free play. The VFU contains centering springs to provide a centering force component, a bobweight effect whereby the input system inertia is altered to provide the pilot with feel forces as a function of normal acceleration, and the normal VFU feel force applied as a function of airspeed (q). Mass balancing of the FCS, except for control surfaces, was required only on the pitch axis FCS. This was accomplished by installing the control column shaker so as to provide the required mass.

Mechanical linkages were used to close the loop in the FCS servoactuators. Figure No. II-5, Elevator System Schematic, and Figure No. 1 (3.2.3.2.3), Aileron Servo Actuator, installation show the input and feedback linkages. Figure No. 1 (3.2.3.2.5.1) shows the inboard elevator surface interconnect linkage which provides a backup for synchronized movement of the two inboard surface segments in the event of a failure. The connections used close tolerance bolts and bearings to minimize free play. In none of the mechanical linkage arrangements did friction and/or free play result in FCS operation below State I.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.1.5 AFCS Design. AFCS shall be provided to the extent specified by the procuring activity.

3.1.5.1 System Requirements. When the specified modes are used, the following design requirements apply.

3.1.5.1.1 Control Stick (or Wheel) Steering. If this mode is required, MIL-F-8785, or if applicable, MIL-F-83300, shall be used as the basis for control capability.

Comparison

The C-5A AFCS was designed to meet the requirement of Contract End Item (CEI) Detail Specification CP 40002-6B.

One of the modes of the AFCS was Control Wheel Steering (CWS) which used the autopilot servos to command a surface position. Therefore, the control capability is dependent upon autopilot capability which is limited by servo authority. The C-5A does not meet Requirement 3.1.5.1.1.

Discussion

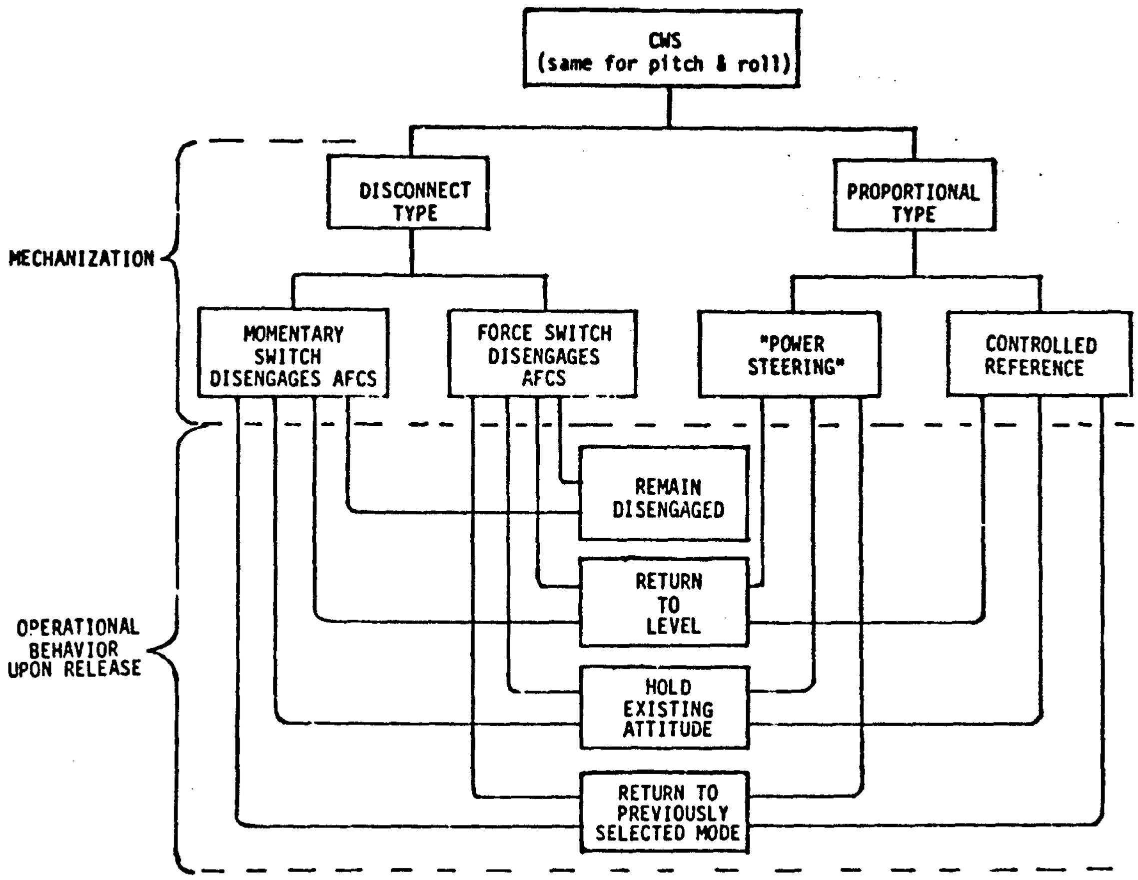
There are fourteen major basic CWS configurations. These configurations are listed in ARINC Report No. 417, dated April 9, 1971, and are given in Figure No. 1 (3.1.5.1.1). The requirement of Paragraph 3.1.5.1.1 to meet the control capability of MIL-F-8785, or if applicable, MIL-F-83300, requires that the pilot have full manual system authority during CWS. This can only be achieved by using the Disconnect Type CWS or by adding additional circuitry to the proportional CWS to produce automatic disengagement and reengagement of the mode. Since the CWS mode is primarily a mode of the autopilot, then it is limited by basic autopilot performance and safety requirements and should not be considered as a separate system which would cause unnecessary and complicated hardware to be added to the system.

This same type of requirement is also contained in Requirement 3.1.2.16 which has been validated and changes suggested (Ref. Rev. 09-21-76).

Recommendation

A. Retain the requirements of 3.1.5 and 3.1.5.1 as stated.

B. Delete the requirement 3.1.5.1.1.



Type	Switch	AFCS Behavior Upon Release of Wheel/Column
Disconnect	Momentary	Remains Disengaged
Disconnect	Force	Remains Disengaged
Disconnect	Momentary	Return to Level
Disconnect	Force	Return to Level
Disconnect	Momentary	Hold Existing Attitude
Disconnect	Force	Hold Existing Attitude
Disconnect	Momentary	Return to Previously Selected Mode
Disconnect	Force	Return to Previously Selected Mode

Type	Reference	AFCS Behavior Upon Release of Wheel/Column
Proportional	Power Steering	Return to Level
Proportional	Power Steering	Hold Existing Attitude
Proportional	Power Steering	Return to Previously Selected Mode
Proportional	Controlled Reference	Return to Level
Proportional	Controlled Reference	Hold Existing Attitude
Proportional	Controlled Reference	Return to Previously Selected Mode

FIGURE NO. 1 (3.1.5.1.1) BASIC CWS CONFIGURATIONS

Requirement

3.1.5.1.2 Flight Director Subsystem. If common mode selection is used, it shall be possible to select control stick steering with flight director operation in place of any of the other AFCS modes. Single-channel flight director operation shall be possible when all except one channel of a redundant system have failed.

Comparison

Two Flight Director Systems (FDS) are installed on the C-5A airplane, System No. 1 for the Pilot and System No. 2 for the Copilot. Each system, Figure 1 (3.1.5.1.2), is comprised of the following LRU's:

- Flight Director Computer (FDS)
- Attitude Director Indicator (ADI)
- Horizontal Situation Indicator (HSI)
- Remote Horizontal Situation Indicator Control Penel (RHSI)
- Peripheral Command Indicator (PCI)
- Navigation Selector Panel (NSP)
- Auxiliary Navigation Select Panel (ANSP)
- Rate of Turn Sensor (RTS)

The FDS provides the integrated display data required for manual instrument flying and for visual monitoring during Automatic Landing approaches and other Autopilot modes. Fifteen different modes of operation are available for use by the FDS, including the FD Self Test mode. These modes are defined as follows:

- Manual Heading (HDG)
- Inertial Heading/Destination Steering (IH/DS)
- Visual Omni Range (VOR): Cruise (CRS) and Approach (APP) Configurations
- Tactical Air Navigation System (TACAN): Cruise (CRS) and Approach (APP) Configurations
- Station Passage (SP) (Associated with the VOR and TACAN modes)
- Instrument Landing System (ILS): (Available from two independent sources, ILS-1 or ILS-2)
- Course Line (CL), also known as Track Steer (Available from two independent sources - Primary Guidance Computer or Auxiliary Guidance Computer)
- Air Drop (AD) also known as terminal navigation (TN) (available from two independent sources - Primary Guidance Computer or Auxiliary Guidance Computer)
- Vertical Navigation (VN) (Available from the Primary Computer only)
- Radar Approach (RA)
- Terrain Following
- Go-Around (GA)
- Altitude Hold (AH) (Available from CADC No. 2 only)
- Navigation Aids Off (Nav. Aids Off)
- Flight Director Self Test (FD ST)

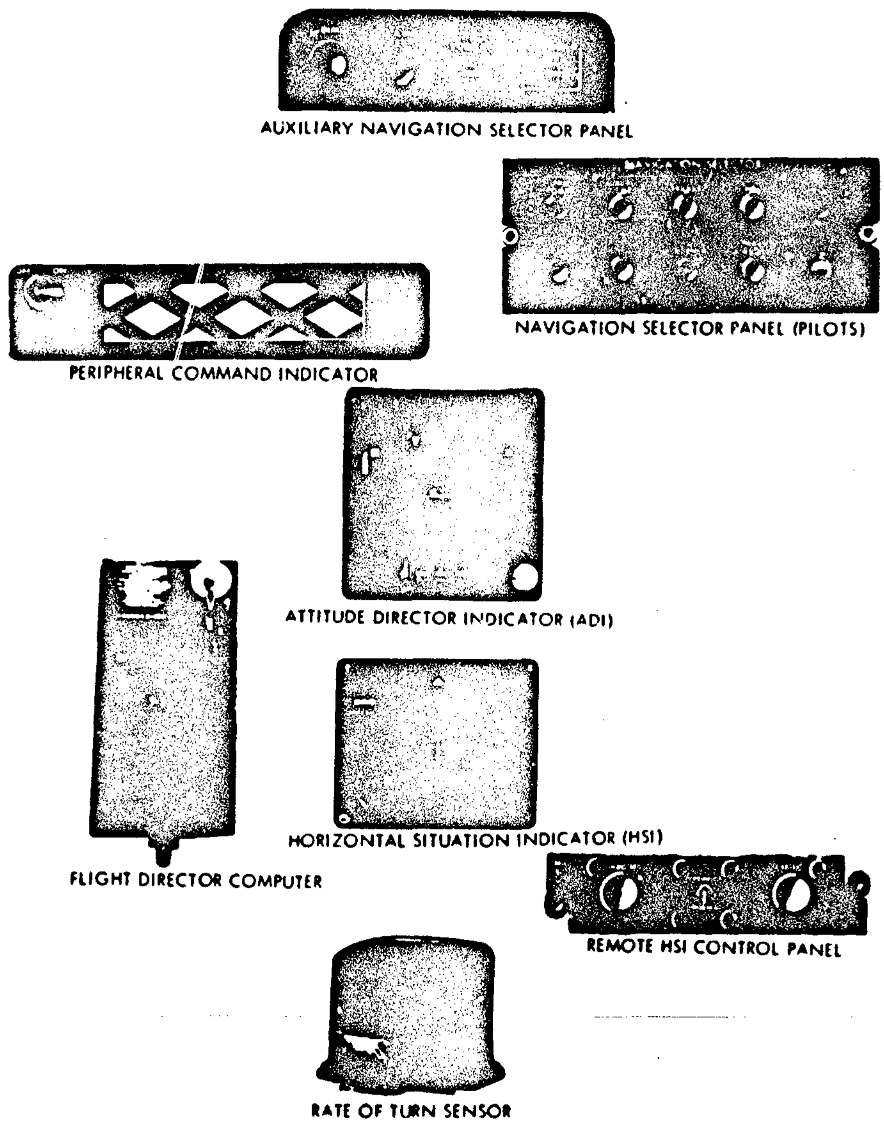


FIGURE 1 (3.1.5.1.2) FLIGHT DIRECTOR EQUIPMENT

The pilot or copilot can independently select the mode he desires by operating the respective mode select switch(es) on his NSP and when applicable, the ANSP. When the autopilot is engaged the CWS mode can be used with flight director operation independent from any other mode of the autopilot.

All the control functions that are provided by the automatic systems are displayed to the pilots via the flight directors for control and monitoring. At any time during any automatic mode of operation, the pilot can take over control of the aircraft by disengagement or override of the automatic system.

It is felt that the C-5A FCS meets the intent of this requirement.

Discussion

Paragraph 1.1 of MIL-F-9490D states that this specification covers "dedicated displays." Paragraph 1.2.1.2 which puts forth the definition of AFCS states that "semiautomatic flight path" control equipment is included. The Background Information and User's Guide for MIL-F-9490D, in the discussion on Paragraph 1.2.1.2, states "Semiautomatic control includes flight director function when the option of automatic or semiautomatic operation is provided." From these statements, it is concluded that this specification, MIL-F-9490D, covers Flight Director and Flight Instruments (See Validation of Paragraph 1.2.1.2). The only requirement that can be found which covers flight director design is this paragraph (3.1.5.1.2) which is inadequate for the basic requirement. The second sentence of this requirement is confusing since it discusses redundant channels of flight directors which is only one of the design approaches that can be made to meet this statement. The pilot must have a triple redundant system so that if one channel fails, then the known working channel is left.

To date, the pilot and copilot flight director systems have been kept completely independent, such that if one failed then the other could continue control. It is felt that more basic design information on the Flight Director System, which would include flight instruments, must be given. All paragraphs that discuss modes of the AFCS should contain a statement defining the interface or dependence on the Flight Director System. Some of these modes are approach, landing, auto-land, and go-around.

Recommendation

Retain the requirement as stated until the recommended study is completed. Perform a study to determine the extent that these requirements should be expanded in order to adequately give the minimum design requirements for the Flight Director System as well as to define the changes to other relative paragraphs of MIL-F-9490D. Areas that could be covered by the new requirement are:

- Flight Direction System Redundancy - Pilot and copilot separation

- Flight Instruments - General requirements on the instruments to be used.

- Transfer Capability - Capability of the pilot or copilot to use signals from the other flight director computer to be displayed on the flight instruments.

- Modes - Modes that could be supplied with same basic requirements Reference can be given to the discussion given for the automatic modes.

- Failure Annunciation - Requirement on annunciator for failures within each system.

- Control Panel - Requirement for independent and shared control panels.

- Integrated Autopilot and Flight Director Requirement - This requirement should define the extent the autopilot functions can be contained within the flight director. A great majority of new AFCS have integrated these two systems.

Additional Data (For "Users' Guide)

The sentences indicated by the left vertical sideline should be added to the background information and "Users' Guide."

Requirement

3.1.5.2 AFCS Interface

3.1.5.2.1 Tie-in with External Guidance. Internal FCS switching with zero command signal input from external guidance systems shall not cause transients greater than engage transients in accordance with 3.1.5.2.3. Noise content in usable external guidance signals shall not saturate or bias any component of the FCS, shall not impair the response of the aircraft to the proper guidance signals, and shall not cause objectionable control motion or attitude variation. Steering information transmitted to the AFCS shall be compatible with the accuracy and dynamic performance requirements of the guidance loop. The tie-in provisions shall not degrade performance of other subsystems by causing excessive loading or saturation.

Comparison

The external guidance sources for the C-5A AFCS interface the autopilot only. The Go-Around Attitude Subsystem, Stallimiter, Autothrottle, Augmentation Systems, and ALDCS have no external guidance inputs. The pitch autopilot receives guidance data from the Primary Navigation Computer, the Multimode Radar Subsystem, and both Glideslope receivers. The roll autopilot receives information from the VOR/localizer receivers, the flight directors, TACAN receiver and navigation system through the pilot navigation select panel. These signals' tie-in with the AFCS was required to meet requirements in CEI Specification CP 40002-6B. Those requirements specified that the reference and command signals be based on the same voltage source as the corresponding feedback signal of the AFCS. Additionally, the command signals from the external guidance systems are required to be limited so that the AFCS will not cause the air vehicle to exceed maneuver limits that are inconsistent with the external guidance function and flight condition.

CP 40002-6B requires that the switching transients with zero command signals from the external guidance systems be less than ± 0.05 g normal acceleration (c.g.) in pitch or ± 1 degree in roll attitude. Finally, the noise content in the external guidance signal is specified so as not to cause improper operation due to saturation of any AFCS component, impair the response of the aircraft to the proper guidance command, and not cause objectionable control motion or attitude variation.

Formal in-flight verification testing of these requirements was not conducted, but performance was informally verified by the many occasions on which the guidance systems were coupled to the AFCS during development testing. On these occasions no deleterious transients, noise content, etc., were experienced.

The mechanism used to establish interface control throughout the C-5A AFCS was the Interface Control Document. Examples of the interface sheets contained in this document are shown in Figure 1 (3.1.5.2.1) (five sheets). These sheets were compiled for each AFCS interface. It allowed an orderly

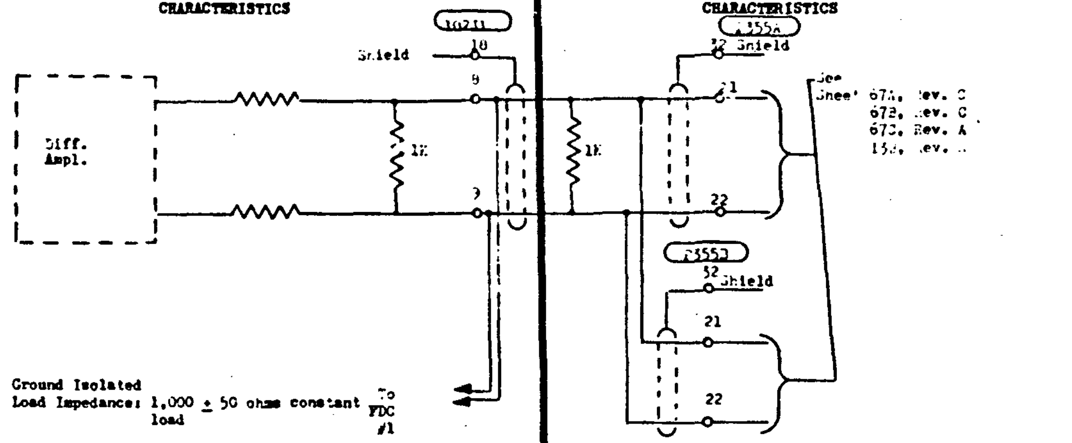
C-5A INTERFACE SHEET **REVISION NO.** N/C **DATED** 5/12/70

NAME OF SIGNAL	TYPE OF SIGNAL	FUNCTION OF SIGNAL
Radar Approach #1	DC Analog	Control Signal for Automatic Radar Approach Mode

UNIT	PLUG NO.	PIN NO.	UNIT	PLUG NO.	PIN NO.
Pitch/Pacs Autopilot Computer 3349?B	P521A-A	J1-8,9,10	MN Radar	P355A P3553	-21,-22,-32 -21,-22,-32

CHARACTERISTICS

CHARACTERISTICS



Ground Isolated
Load Impedance: 1,000 ± 50 ohms constant load

APPROVALS

NAME	NAME	72-13	72-19
COMPANY	COMPANY	72-	72-15

FIGURE 1 (3.1.5.2.1). C-5A EXAMPLE AFCS INTERFACE (SHEET 1)

ICA C-19 Rev. N/C — ICA C-18A Rev. A — ICA C-18B Rev. B — ICA C-565 Rev. C

C-5 INTERFACE SIGNAL SPECIFICATION — Section C — SHEET 67B OF

SIGNAL CODE	NAME	TO	FROM	TYPE	FUNCTION/DESCRIPTION
CF-J-4	RADAR GLIDESLOPE DEVIATION 1	FLIGHT STATION	MMR X DPU	ANALOG DC	

OUTPUT DEVICE: Resistive divider (Z_o = 150 ohm max) via X DPU Relay Contacts. Contacts are closed only when X band radar is operating in RA mode. Max contact rating = 250 ma resistive.

NOTE: Output amplitude not to exceed 1 vdc as instrument safety precaution.

PHASING AND SIGN CONVENTION: Positive Voltage = Aircraft above set glideslope
Negative Voltage = Aircraft below set glideslope

INDEX REFERENCE: 0 vdc = 0°

LOAD DEVICE:

Autopilot:	1 K ohm ± 5%
FDC 1:	1 K ohm ± 5%
ADI 1:	1 K ohm ± 5%
Dummy:	1 K ohm ± 5%

Constant 250 ohm ± 5% load must be maintained and isolated from FDC, ADI and Autopilot grounds.

MAXIMUM ALLOWABLE SIGNAL NOISE: TBD

INTERCONNECTION: This output is externally junctioned with signal code CE-1-4 and then externally branched at a junction panel to FDC #1 and the Autopilot Pitch Computer. Signal goes through FDC #1 to ADI #1 load. Ref. Figure. C-12 & C-37.

WIRING CATEGORY: 4

FUNCTION: Signal sent to Autopilot, Flight Director Computer and instrument to provide Radar Approach to Landing capability automatically and/or manually.

DESCRIPTION: Signal generated in the MMR glideslope computer after relating preset glideslope angle (Navigation MMR control panel 2° to 7°) and the measured depression angle to the terrain aim point. Signal is linearly proportioned to difference between preset angle and depression angle for a range of ± 1°. Signal to continue (with sign maintained) nonlinearly to maximum 1 vdc limit. Signal allowed to saturate between 0.4 vdc and 1 vdc. Sign of saturated deviation signal to be of correct sense for computed deviations up to ± 15°.

ACCURACY: 15 mv (1σ) for selected glide slope angles of 3° to 7°.

SCALE FACTOR: 150 mv/degree over the range of ± 1°.

FIGURE 1 (3.1.5.2.1). C-5A EXAMPLE AFCS INTERFACE (SHEET 2)

ICA C-18 Rev. N/C | ICA C-18A Rev. A | ICA C-18B Rev. B | ICA C-565 Rev. C

C-5 INTERFACE SIGNAL SPECIFICATION Section C SHEET 67A

SIGNAL CODE	NAME	TO	FROM	TYPE	FUNCTION DESCRIPTION
CE-J-4	RADAR GLIDESLOPE DEVIATION 1	FLIGHT STATION	MMR K_u DPU	ANALOG DC	

<u>OUTPUT DEVICE</u>: Resistive divider (Z_o = 150 ohm max) via K_u DPU

relay contacts. Contacts are closed only when K_u band radar is operating in RA mode. Max contact rating = 250 ma resistive.

<u>NOTE</u>: Output amplitude not to exceed 1 vdc as instrument safety precaution.

<u>PHASING AND SIGN CONVENTION</u>: Positive Voltage = Aircraft above set glideslope
Negative Voltage = Aircraft below set glideslope

<u>INDEX REFERENCE</u>: 0 vdc = 0°

<u>LOAD DEVICE</u>:

Autopilot:	1 K ohm ± 5%
FDC 1:	1 K ohm ± 5%
ADI 1:	1 K ohm ± 5%
Dummy:	1 K ohm ± 5%

Constant 250 ohm ± 5% load must be maintained and isolated from FDC, ADI and Autopilot grounds.

<u>MAXIMUM ALLOWABLE SIGNAL NOISE</u>: TBD

<u>INTERCONNECTION</u>: This output is externally junctioned with signal code CF-J-4 and then externally branched at a junction panel to FDC #1 and the Autopilot Pitch Computer. Signal goes through FDC #1 to ADI #1 load. Ref. Figures C-12 & C-37.

<u>WIRING CATEGORY</u>: 4

<u>FUNCTION</u>: Signal sent to Autopilot, Flight Director Computer and instrument to provide Radar Approach to Landing capability automatically and/or manually.

<u>DESCRIPTION</u>: Signal generated in the MMR Glideslope Computer after relating preset Glideslope angle (Navigation MMR control panel 2° to 7°) and the measured depression angle to the terrain aim point. Signal is linearly proportioned to difference between preset angle and depression angle for a range of ± 1°. Signal to continue (with sign maintained) nonlinearly to maximum 1 vdc limit. Signal allowed to saturate between 0.4 vdc and 1 vdc. Sign of saturated deviation signal to be of correct sense for computed deviations up to ± 15°.

<u>ACCURACY</u>: 15 min (1σ) for selected glideslope angles of 3° to 7°.

<u>SCALE FACTOR</u>: 150 mv/degree over the range of ± 1°.

FIGURE 1 (3.1.5.2.1). C-5A EXAMPLE AFCS INTERFACE (SHEET 3)

I. C-565
REV A

C-5 INTERFACE SIGNAL SPECIFICATION Section C SHEET 67C OF REVISION A DATE 3/21/69

SIGNAL CODE	NAME	TO	FROM	TYPE	FUNCTION/DESCRIPTION
CE & CF-J-4	RADAR GLIDESLOPE DEVIATION-1	FDC-1 & AFCS	MMR-X-& Ku-DPU	DC ANALOG	

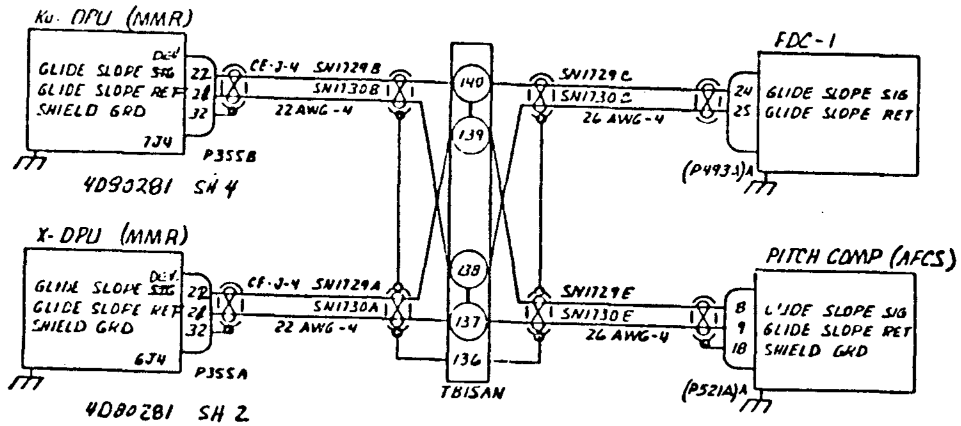
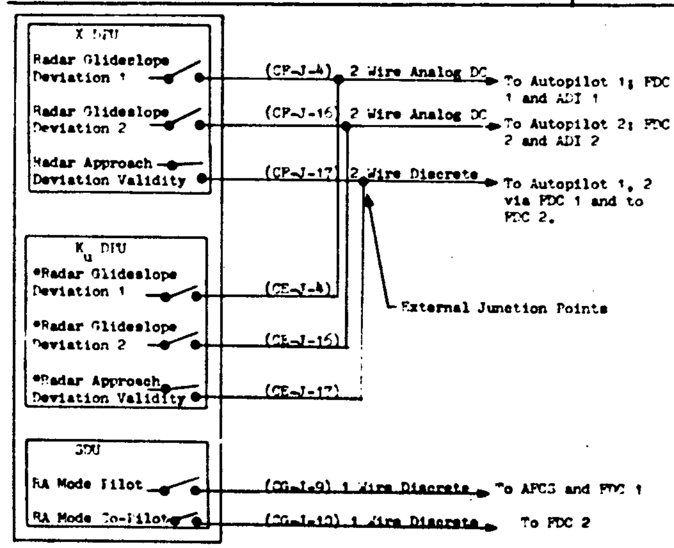


FIGURE C-37

FIGURE 1 (3.1.5.2.1). C-5A EXAMPLE AFCS INTERFACE (SHEET 4)

Signal Code	Name	To	From	Type	Function / Description
	FIGURE C-12 RADAR APPROACH MODE	FLIGHT STATION	MMR	ANALOG DC, DISCRETE	



*These outputs normally supplied from this DPU. If this DPU fails or operator elects to change radars for any operational reason, the output will be supplied by other DPU.

NOTE: Outputs are not switched automatically to the other radar in the event of a radar malfunction. Switching is performed manually.

FIGURE 1 (3.1.5.2.1). C-5A EXAMPLE APCS INTERFACE (SHEET 5)

approach to interface design when more than one equipment vendor was involved.

Discussion

The C-5A complies with this requirement. Compliance can be practically demonstrated and the requirement is suitable to future transport aircraft. It is felt that no changes are necessary for this requirement.

Recommendation

Accept "as is."

Requirement

3.1.5.2.2 Servo Engage Interlocks. Interlocks shall be provided to prevent servo engagement and to provide disengagement in the presence of conditions that render disengagement safer than engagement. Manual override of interlocks shall be provided wherever such override capability will enhance flight safety.

Comparison

Those elements of the C-5A AFCS which use servo engage interlocks include the pitch and roll autopilots, the pitch and yaw/lateral augmentation systems. CEI Specification CP 40002-6B specifies the following:

> "Interlocks to prevent engagement of the AFCS in the absence of electrical power of the proper voltage and frequency, proper gyro rotor speed, adequate warmup, and normal overall operation shall be provided.... It shall not be possible to engage incompatible functions. Interlocks shall be provided to prevent power from being applied to the engaged mechanisms if lack of power to the servo unit prevents synchronization. In the event of failure of any one of the power sources, the AFCS shall become disengaged within 0.3 second."

The pitch autopilot can be engaged independently of the roll autopilot to provide split axis operation. Figure 1 (3.1.5.2.2) shows that the computer does contain a number of interlocks to prevent engagement if any interlock is not valid. Similar interlocks are used for the roll autopilot servo.

The engage/disengage circuitry for the pitch and yaw/lateral augmentation systems is enabled by the fault detection/correction logic which assures that these subsystems cannot be engaged into an uncorrected fault situation. Interlocks are also provided the self-test initiation of these systems (pitch and yaw/lateral augmentation) to assure that the test sequence cannot be initiated in-flight or on the ground with the control surfaces powered. These interlocks are provided by the touchdown relays and aileron, rudder, and inboard elevator hydraulic power switches on the overhead flight station control panel see Figure 2 (3.1.5.2.2). This figure shows the BITE initiate power is routed from the computer through the Flight Augmentation Control Panel to two touchdown relays. If weight is on the landing gear, the relays close and supply power to the BITE push-button on the computer face and the TEST ON switch in the panel. Momentary depression of either switch will begin "BITE Initiate." Because the signals are momentary, the BITE Initiate circuitry must "Latch-on" to begin the test sequence. This latching power is routed through the surface hydraulic switches on the overhead panel and all switches must be in the "OFF" position to complete the interlock and accomplish BITE latch.

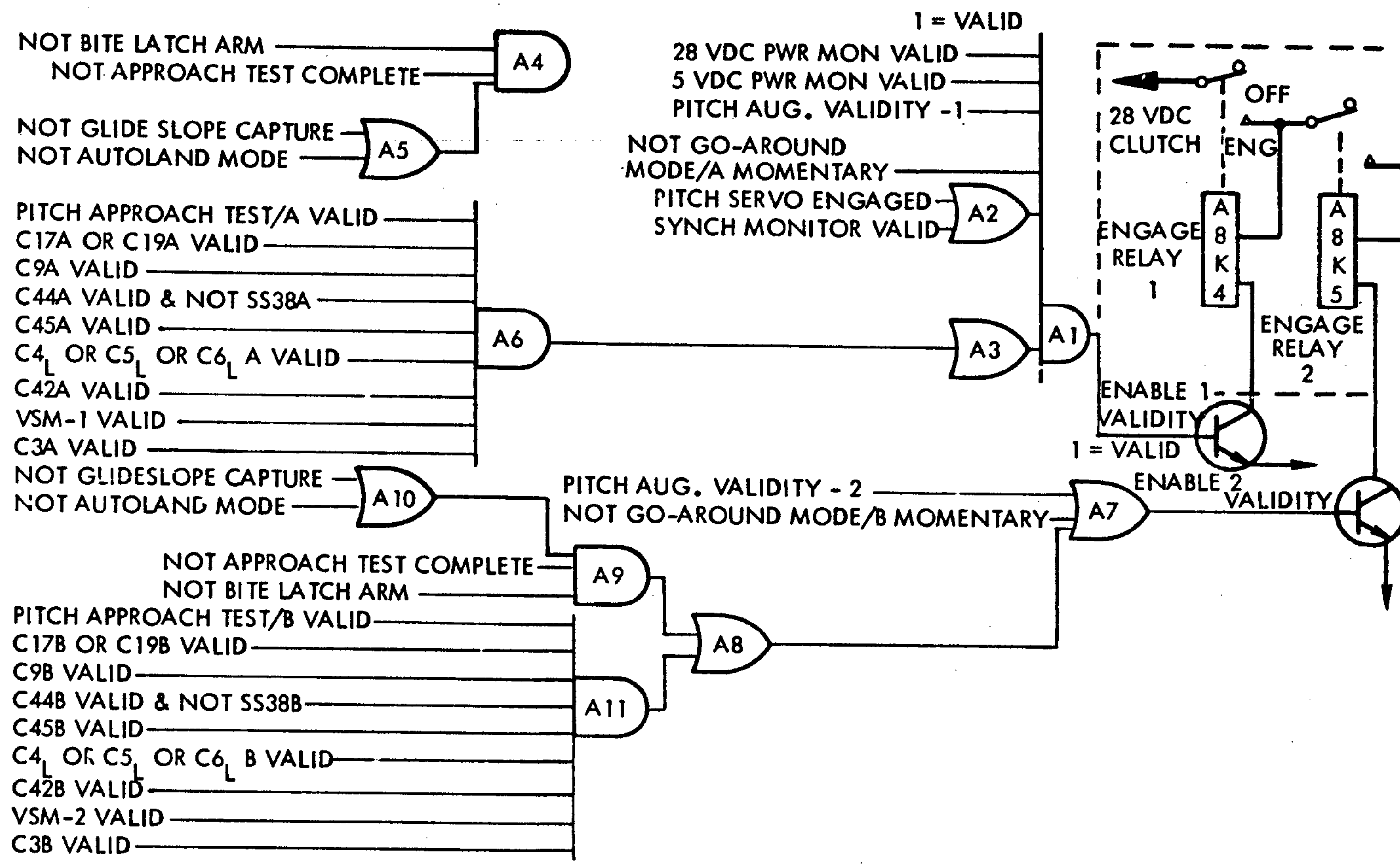


FIGURE 1 (3.1.5.2.2). PITCH AUTOPILOT SIMPLIFIED ENABLE LOGIC

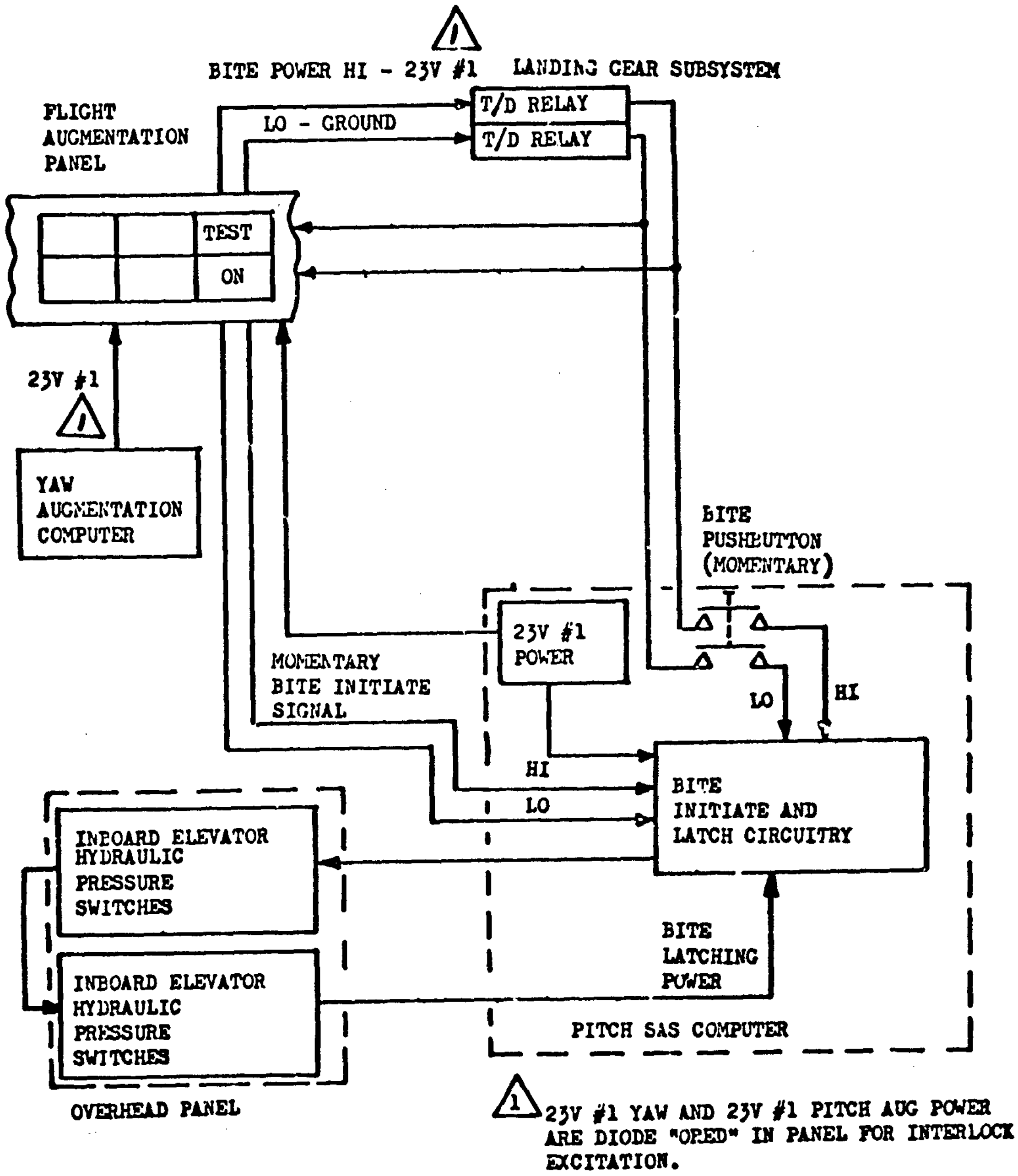


FIGURE NO. 2 (3.1.5.2.2) PITCH AUGMENTATION SELF TEST INTERLOCK SCHEMATIC

No manual override capability is provided in any of the above interlocked systems.

Discussion

A rational approach to the use of servo engage interlocks involves an assessment of conditions--both system internal operating conditions and external flight operating conditions to determine the AFCS function performance provided through the servo and to assess resultant aircraft safety. Those conditions which render disengagement safer than engagement will require automatic servo interlocks. Likewise, those conditions which render servo engagement safer than disengagement will be identified by such an analysis. Thus, properly designed and implemented servo interlocks would not require the additional complexity of manual override provisions or the attendant additional analytical task of determining for each interlock circuit whether an additional manual override would or would not enhance flight safety.

Recommendation

Delete the last sentence of the requirement.

Requirement

3.1.5.2.3 Engage-Disengage Transients. Normal engagement or disengagement of AFCS modes shall not result in transients exceeding the limits set on MFCS engage-disengage transients by MIL-F-8785 and MIL-F-83300. Normal engagement transient requirements shall be met 2 seconds after completion of any maneuver up to the maneuver limits of the aircraft or the limits of sensor equipment being used.

Comparison

Since the C-5A is a CTOL transport, MIL-F-83300 is not applicable. The MIL-F-8785B requirements for normal MFCS engagement-disengagement are that the transients not exceed ± 0.05 g normal or lateral acceleration at the pilot's station and ± 1 degree per second roll rate when within the Operational Flight Envelope and that the transients not exceed ± 0.5 g normal or lateral acceleration at the pilot's station, ± 5 degrees per second roll rate and the lesser of ± 5 degrees sideslip or the structural limit when within the Service Flight Envelope. The C-5A design specification, CP 40002-6B, sets tighter limits on the normal engagement-disengagement transients requiring normal accelerations at the cg not to exceed ± 0.05 g and roll attitude changes of not more than ± 1 degree.

The C-5A Category II flight test evaluation, Report FTC-TR-73-41 "shows that there were no transients during A/P engagement or disengagement tests." Therefore, the C-5A is in compliance with this requirement.

Discussion

This requirement is well stated and valid. Compliance can be practically demonstrated as it was for the C-5A. The statement of the requirement provides continuity with the handling qualities military specifications. The paragraph is applicable to future transport design.

Recommendation

Retain the requirement as stated.

Requirement

3.1.5.3 AFCS Emergency Provisions

3.1.5.3.1 Manual Override Capability. It shall be possible to manually overpower or countermand the automatic control action of the AFCS using the normal pilot controls. Required pilot forces shall not exceed pilot capabilities as defined by MIL-STD-1472. The overpower force for V/STOL aircraft and helicopters shall not exceed the limit cockpit control forces specified for Level 1 operation in MIL-F-83300. Manually overriding the AFCS shall not result in an instability due to force flight between the pilot and the AFCS.

3.1.5.3.2 Emergency Disengagement. Positive emergency means of disengagement, in addition to normal mode selection, shall be provided for AFCS. The emergency disengagement means shall also ground the power input side of the servo engage solenoids. No intervening switching mechanism between the point of ground and the solenoid shall exist.

Comparison

The C-5A is designed with the capability of manually overpowering the AFCS using the normal pilot controls. The maximum overpower forces applied at the pilot's controls must be less than 120 pounds rudder, 50 pounds elevator, and 40 pounds aileron. These forces are within the pilot's capabilities as defined by MIL-STD-1472.

The AFCS modes can be manually disengaged by pressing the appropriate engage switch a second time, by depressing either control wheel's disengage switch, or by switching the MASTER POWER off. Automatic disengagement occurs when a failure is detected by the monitoring system and appropriate annunciation provided for the pilots.

Discussion

The C-5A complies with the above requirements which can be practically demonstrated. The requirements are straightforward and applicable to transport aircraft. The requirements are appropriate as stated and will not require changes.

Recommendation

Accept "as is."

Requirement

3.1.6 Mission Accomplishment Reliability. The probability of mission failure per flight due to relevant material failures in the flight control system shall not exceed the applicable limit specified below. Failures in power supplies or other subsystems that do not otherwise cause mission failure shall be considered where pertinent. Each mission to which this requirement applies shall be established and defined by the contractor, subject to approval of the procuring activity.

a. Where overall aircraft mission accomplishment reliability is specified by the procurement activity, $Q_{M(fcs)} \leq (1-R_M)\ A_{M(fcs)}$

b. Where overall aircraft mission accomplishment reliability is not specified, $Q_{M(fcs)} \leq 1 \times 10^{-3}$

where: $Q_{M(fcs)}$ = Maximum acceptable mission unreliability due to relevant FCS material failures.

R_M = Specified overall aircraft mission accomplishment reliability.

$A_{M(fcs)}$ = Mission accomplishment allocation factor for flight control (chosen by the contractor)

Comparison

The C-5A had a contractual overall aircraft mission reliability requirement (R_M) of 98% (ref. paragraphs 3.13 and 3.14 of C-5 Reliability Program Plan, Document 3-8) based on a contractual set of abort criteria and a ten hour flight length. Achievement of this requirement was verified in the 1973 C-5 Operational Reliability Verification Program. Failure data collected from 25 C-5A aircraft during the period January through June 1973 was utilized in the verification. The flight control system was allocated 12.5% of the total mission failure requirement.

The FCS accounted for 9.2% of the total observed mission failures. The mission unreliability observed in the verification program for the C-5A flight control system was .0045. This value corresponds to the $Q_{M(fcs)}$ factor used in parts a and b of paragraph 3.1.6. The $Q_{M(fcs)}$ relationship defined in the part a equation is a valid one for situations where overall aircraft mission reliability requirements are specified by the procurement activity. However, the quantitative requirement described in part b ($Q_{M(fcs)} \leq 1 \times 10^{-3}$) is not met by the C-5A and is considered to be unrealistic and in need of revision.

Discussion

Even though the quantitative value of part b is supposed to apply only when the overall aircraft mission reliability is not specified, the fact that a specific $Q_{M(fcs)}$ value is stated establishes it as a target or guideline for

future contractually negotiated values. As it presently reads, part b stipulates that $Q_{M(fcs)} \leq .001$ with no adjustment provisions for flight length. It is obviously unreasonable to expect the unreliability levels for one hour and ten hour flights to be the same. The C-5A, for example, with its ten hour flight length has a mission unreliability 4.5 times the proposed MIL-F-9490D requirement. For small failure rates mission unreliability (Q_M) is equal to λt where λ is the hourly failure rate and t is the aircraft flight length. Since flight length is an influential part of the mission unreliability calculation and since flight length varies for different aircraft, it would be more appropriate to specify a quantitative failure rate objective in part b rather than a quantitative Q_M. To substantiate this viewpoint recent Air Force 66-1 operational data was examined for the C-5A, C-141 and C-130E/H aircraft. $Q_{M(fcs)}$ values, flight lengths, and failure rates developed from this data are summarized below along with the C-5A verification program values. The C-5A verification program failure rate which was developed from assessed data including relative failures only per defined abort criteria is understandably lower than the C-5A 66-1 raw data failure rate. Also, as you would expect, the C-5A 66-1 data failure rate is higher than the failure rates for the less complex C-141 and C-130E/H.

	$Q_{M(fcs)}$	Flight Length(Hrs.)	Failure Rate(Aborts/FH)
C-5A (Verification Program)	.00450	10.0	.00045
C-5A (66-1 Data)	.00302	4.5	.00067
C-130E/H	.00112	2.8	.00040
C-141	.00092	3.4	.00027

The C-141 with its 3.4 hour flight length meets the $Q_{M(fcs)}$ requirement of paragraph 3.1.6 part b and the C-130E/H with its 2.8 hour flight length nearly does. Note that even though the C-5A verification program failure rate and the C-130E/H failure rate are essentially the same, the flight length difference causes the $Q_{M(fcs)}$ values to vary substantially. This chart clearly illustrates the impact of flight length on $Q_{M(fcs)}$.

It is recommended that part b be revised to specify that $Q_{M(fcs)} \leq \lambda t$ where λ is the flight control system failure rate based on mission reliability criteria and t is the aircraft flight length. Based on data examined for the C-5A, C-141, and C-130E/H and in view of the fact that future aircraft flight control systems will probably be at least as complex as the C-5A, a failure rate equal to .0005/FH is proposed. Part b would therefore read $Q_{M(fcs)}$ .0005t. For a one hour flight $Q_{M(fcs)} \leq .0005$, for a four hour flight $Q_{M(fcs)} \leq .002$, and so on.

Comparison of the proposed requirement with the present requirement shows that t would be equal to 2.0 hours for a $Q_{M(fcs)} = .001$. This indicates that the present requirement is reasonable for flight lengths of 2.0 hours or less and too stringent for longer flight lengths. Otherwise this is a valid requirement which has been satisfied by the C-5A FCS design to the extent noted. The requirement can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Revise the requirement as follows:

Revise b. to read:

b. Where overall aircraft mission accomplishment reliability is not specified, $Q_{M(fcs)} \leq .0005t$ where t = aircraft flight length. (Retain existing definitions of $Q_{M(fcs)}$, R_M, and $A_{M(fcs)}$).

Requirement

3.1.7 Quantitative Flight Safety. The probability of aircraft loss per flight, defined as extremely remote, due to relevant material failures in the flight control system shall not exceed:

$$Q_{S(fcs)} \quad (1-R_S)\ A_{S(fcs)}$$

where $Q_{S(fcs)}$ = Maximum acceptable aircraft loss rate due to relevant FCS material failures.

$A_{S(fcs)}$ = Flight safety allocation factor for flight control (chosen by the contractor).

= Overall Aircraft Flight Safety Requirements as specified by the procuring activity.

Failures in power supplies or other subsystems that do not otherwise cause aircraft loss shall be considered where pertinent. A representative mission to which this requirement applies shall be established and defined in the FCS specification (4.4.2). If overall aircraft flight safety in terms of R_S is not specified by the procuring activity, the numerical requirements of Table VII apply.

TABLE VII
FCS QUANTITATIVE FLIGHT SAFETY REQUIREMENTS

		MAXIMUM AIRCRAFT LOSS RATE FROM FCS FAILURES
OVERALL A/C FLIGHT SAFETY REQUIREMENT NOT SPECIFIED BY PROCURING ACTIVITY	MIL-F-8785 CLASS III AIRCRAFT	$Q_{S(fcs)} \leq 5 \times 10^{-7}$
	ALL ROTARY WING AIRCRAFT	$Q_{S(fcs)} \leq 25 \times 10^{-7}$
	MIL-F-8785 CLASS I, II & IV AIRCRAFT	$Q_{S(fcs)} \leq 100 \times 10^{-7}$

Comparison

The C-5A program had no contractual quantitative flight safety requirement. The Lockheed System Safety Program Plan that was contractually approved at the outset of the C-5A program included provisions for a quantitative safety model. There was no required or target quantitative value assigned to the aircraft or flight control system as a result of using this model.

Development and use of the model was discontinued later in the C-5A program by mutual Lockheed/Air Force agreement. As a result of this, no allocated or predicted quantitative values are available for comparison purposes. Actual operational experience for the C-5A aircraft can be used to compare the MIL-F-9490D requirement with actual observed C-5A FCS performance.

This comparison is as follows:

MIL-F-9490D $\quad Q_{S(fcs)} \leq 5 \times 10^{-7}$ loss rate per mission

C-5A Actual $\quad Q_{S(fcs)} = 0.0$ loss rate per mission

It is obvious that the C-5A would have met the requirement for quantitative flight safety if the MIL-F-9490D maximum value had been imposed.

Substantiating data for the above comparison is as follows:

- C-5A Flight Hours cumulative through 30 August 1976 = 256,633 Flt. Hrs.
- Average C-5A mission length = 4.5 Flt. Hrs.
- Total C-5A losses (relevant FCS failures) = 0
- C-5A FCS <u>hourly</u> loss rate $\dfrac{0 \text{ (FCS losses)}}{256{,}633 \text{ (flight hours)}} =$

  $Q_{S(fcs)} = 0.0$ (hourly rate)
- C-5A FCS <u>mission</u> loss rate

  $$\frac{0 \text{ (FCS losses)}}{256{,}633 \text{ (flight hours)}} \times 4.5 \text{ (avg. mission length)} =$$

  $Q_{S(fcs)} = 0.0$ (mission rate)

<u>Discussion</u>

Table VII establishes the maximum aircraft loss rate from FCS failures as $Q_{S(fcs)} \leq 5 \times 10^{-7}$ for heavy cargo type aircraft. This value is imposed on the contractor if overall aircraft flight safety, in terms of R_S, is not specified by the procuring activity. The implication of this MIL-F-9490D requirement is to suggest that the $Q_{S(fcs)}$ value of 5×10^{-7} should be a goal or target that is achievable in most cases. It also implies that the $(1-R_S)$ $A_{S(fcs)}$ value should not exceed 5×10^{-7}. The conclusion based on the C-5A comparison would be that this $Q_{S(fcs)}$ value is acceptable.

It should be noted that the $Q_{S(fcs)}$ value is an aircraft loss rate per flight which imposes a requirement that is independent of mission length. Hardware failures and the resulting failure rates that lead to Category IV catastrophic loss of aircraft are dependent on mission length. Therefore, a quantitative requirement by MIL-F-9490D, independent of average mission length, is not

valid. MIL-F-9490D does state that a representative mission shall be defined by the FCS specification originated by the contractor. This will not resolve the problem pointed out above since the defined mission and average mission length, even though realistic, would not in many cases be compatible with the $Q_{S(fcs)}$ required value.

The C-141 was evaluated in the same manner that the C-5A was for comparison purposes. The results of this analysis reveal that the C-141 would have also met the MIL-F-9490D requirement. This is based on the following:

- C-141 cumulative flight hours through September 1976 = 4,730,000 hours
- Average C-141 mission length = 3.8 hours
- Total C-141 losses (relevant FCS failures) = 0
- C-141 FCS loss rate = 0

Recommendation

It is recommended that Table VII be revised to reflect an aircraft loss rate that is a function of mission length. The entry for column three of the table for Class III aircraft should be:

$$Q_{S(fcs)} \leq (\lambda) \quad (t)$$

where (λ) is the flight control system average hourly loss rate based on the representative mission definition and failure criteria. This value should be $\leq 3 \times 10^{-7}$.

(t) is the average mission length for the representative mission defined in the contractor's FCS specification or by the procuring activity.

Derivation of the λ value is based on data contained in Air Force Flight Dynamics Laboratory Technical Report 74-116, Background Information, and Users' Guide for MIL-F-9490D. Appendix A of that report contains actual field safety experience data for the B-52, C-135, and C-141 aircraft. C-5A data was not added to this data sample for the following reasons:

- 256,633 flight hours of experience for the C-5A is a relatively small data sample compared to the experience data for the B-52, C-135, and C-141.
- There were no C-5A losses due to FCS relevant failures.

Referring to page 219 of Appendix A in TR-74-116 the following data elements were extracted for analysis purposes:

Combined B-52/C-135/C-141 Experience

Flight hours	=	12,248,946
Flights	=	1,826,908
A/C losses (due to relevant FCS failures)	=	1
A/C loss rate per 100,000 flt. hrs.	=	.008
A/C loss rate per 100,000 flights	=	.055

Using the above data it can be seen that the loss of one aircraft (B-52) in 12,248,946 flight hours results in a λ value of .000000081 as an hourly failure or loss rate. This can be considered as the actual observed hourly failure rate. Likewise, the observed mission loss rate is .000000547 or $\frac{1 \text{ loss}}{1{,}826{,}908 \text{ flights}}$.

This value is the MIL-F-9490D requirement expressed as $Q_{S(fcs)} \leq 5 \times 10^{-7}$. It is obvious that this value is based on a questionable data sample, i.e., one aircraft loss. Consequently, it is necessary to evaluate the confidence levels associated with the given data sample. From this, a more realistic requirement value can be obtained with a higher degree of confidence. This is done as follows:

$$\text{Observed } \lambda = \frac{1 \text{ loss}}{12{,}248{,}946 \text{ flt. hours}} = .8 \times 10^{-7}$$

$$\lambda \text{ at 90\% confidence (lower limit)} = 3.17 \times 10^{-7}$$

Developed as follows:

$$\text{MTBF}_{\text{90\% confidence lower limit}} = \frac{2\,(T)}{\chi^2_{2n+2\ \text{(degrees freedom) 90\% lower limit}}}$$

Where: T = flight hours = 12,248,946

χ^2_{2n+2} = Chi-square mathematical distribution for $2n + 2$ degrees freedom

n = aircraft losses (failure quantity)

$2n + 2$ = degrees of freedom

MTBF = $\frac{1}{\lambda}$

MTBF = mean-time-between-failures(losses)

λ = failure(loss) rate

Therefore:

$$\text{MTBF}\ \begin{pmatrix}90\% \\ \text{lower limit}\end{pmatrix} = \frac{(12{,}248{,}946)}{7.78}$$

$$\text{MTBF}\ \begin{pmatrix}90\% \\ \text{lower limit}\end{pmatrix} = 3{,}148{,}829 \text{ flight hours}$$

$$\lambda = .000000317$$

or $$\lambda = 3.17 \times 10^{-7}$$

The result of this calculation is that one can be 90% confident that the actual observed failure (loss) rate (λ) will be $\leq 3.17 \times 10^{-7}$ based on the B-52/C-135/C-141 experience as shown above.

Further, it can be stated that one can be 90% confident that the mean-time-between-aircraft losses due to relevant FCS failures will be 3,148,829 flight hours based on the same experience data.

The conclusion is that the stated MIL-F-9490D requirement of $.8 \times 10^{-7}$ (hourly rate) should be approximately 3×10^{-7} with 90% confidence. Converting this to mission rate values we see:

Present requirement	$= 5 \times 10^{-7}$ mission rate
Recommended requirement if expressed as a mission rate	$= 6.7 \text{ (avg mission time)} \times (3 \times 10^{-7})$ $= 20 \times 10^{-7}$

Requirement

3.1.7.1 Quantitative Flight Safety - All Weather Landing System (AWLS). The average hazard due to the use of the all-weather landing system shall be less than the risk allowed in the contractor's reliability budget for the all-weather landing system. To meet the requirements of 3.1.7, the contractor shall allocate the FCS safety budget among AWLS and other FCS. The specific risk of a hazard due to use of the landing system under an environment limit or operational restriction shall not increase the allowed risk by a factor of more than thirty. An alert height shall be established at an altitude such that, with all systems operative at the alert height, the probability of a hazard occurring during the landing is extremely remote, as defined in 6.6.

Comparison

The C-5A program had no quantitative flight safety requirements for the all-weather landing system (AWLS). A quantitative flight safety mathematical model was developed initially for the C-5A program, but was later discontinued by mutual agreement between Lockheed and the Air Force. In addition, there were no C-5A losses due to relevant FCS failures, therefore, the C-5A would have met the quantitative requirement stipulated in the above paragraph.

Discussion

The most significant criticism of paragraph 3.1.7.1 is one of semantics. The following phrases are examples:

- average hazard
- risk
- specific risk
- allowed risk

The first sentence compares an average hazard value to a risk value. Interpretation of this requirement can be expressed as follows:

$$\frac{\Sigma \text{ (severity factor) } (\lambda)}{\text{total quantity of hazards}} < (1 \times 10^{-3})\ (A_R(\text{AWLS}))$$

where:

severity factor = Hazard Category, e.g.,
IV = CATASTROPHIC
III = CRITICAL
II = MARGINAL
I = NEGLIGIBLE

λ = Probability of hazard occurrence

total quantity of hazards	=	all hazards identified for the AWLS function
(1×10^{-3})	=	reliability budget (maximum value) for FCS
A_R(AWLS)	=	reliability allocation factor for AWLS as a sub-function of FCS.

The above mathematical expression provides that the average risk of a given AWLS hazard be a function of the reliability budget or allocated failure rate for AWLS. It also requires that the average risk involved for the AWLS function be less than the allocated AWLS failure rate. This implies that:

$$\frac{\Sigma \lambda}{\text{total AWLS hazards}} < \frac{(1 \times 10^{-3})\ (A_S(\text{AWLS}))}{4}$$

This expression states, that for the worst case (Category IV hazards), the average probability of occurrence of a hazard must not exceed 1/4 of the allocated AWLS failure rate. This would presuppose the concept that each 4th AWLS failure would be catastrophic resulting in loss of the aircraft.

The definition of average risk of a hazard is confusing and misleading. Deleting the phrase "average hazard" and inserting "summation of risks of hazards" would clarify this requirement. Difficulty exists when comparisons are attempted between the terminology "average hazard" and "reliability budget." Hazards and the risks associated with them can better be defined and discussed in terms of safety budget, not reliability budget. The desired mathematical expression for this first sentence would be:

Σ (risks of AWLS hazards) $(A_S(\text{AWLS}))$ (4)

or (severity factor) (λ) $(A_S(\text{AWLS}))$ (4)

where:	severity factor	=	hazard category
	λ	=	probability of hazard occurrence
	$(A_S$(AWLS)	=	safety allocation for AWLS
	4	=	severity factor "catastrophic" for AWLS

The above interpretation is believed to be inconsistent with the intent of the first sentence of paragraph 3.1.7.1. The most desirable wording for this sentence should be stated so as to ensure that the probability of normal hazard occurrence during an AWLS landing is less than the allocated probability of hazard occurrence for the $\text{FCS}_{(\text{AWLS})}$ function.

The normal system safety connotation of "risk" is the combination of both probability of hazard occurrence and severity of hazard occurrence.

Therefore the synonymous use in paragraph 3.1.7.1 of hazard probability of occurrence and risk of a hazard is in error. Risk implies severity as well as probability of occurrence.

The definition of average hazard as described in AFFDL-TR-74-116 is a hazard that occurs under normal operating conditions. Normal meaning within the design and operational limitations established for that system. Specific risk pertains to "average" hazards that might occur when operating beyond the design or operational limitations established for that system. Clarification of these two phrases should be included in MIL-F-9490D.

The second sentence of paragraph 3.1.7.1 simply requires the allocation of the safety budget for FCS and AWLS. A quantitative safety allocation can be accomplished by a defined process that would be developed prior to contractual go-ahead.

The third sentence of paragraph 3.1.7.1 deals with the quantitative limitation for an AWLS specific hazard risk. The interpretation is, that for a specific risk, the increase cannot be exceeded by a factor of 30 if the AWLS system is used beyond its designed environmental limits or operational capabilities. The risk in this case is interpreted to mean severity of a hazard multiplied by the probability of occurrence.

The fourth sentence of paragraph 3.1.7.1 provides that the loss rate of the aircraft shall not exceed (5×10^{-7}) once a decision to land has been made. This assumes all equipment working properly at decision height.

The loss rate must be considered in a mission time frame. The AWLS mission or phase of flight is small compared to the overall mission length. The time period between decision height and touchdown is a matter of 10 to 15 seconds, thus, reducing the probability of hazard occurrence if the hazard rate is used as a function of time.

Recommendation

It is recommended that the requirement set forth in paragraph 3.1.7.1 be changed to the following:

> "3.1.7.1 Quantitative Flight Safety - All Weather Landing System (AWLS). The probability of hazard occurrence due to the use of the all-weather landing system under normal conditions including limited visibility shall be less than the probability allowed in the contractor's safety budget for the all-weather landing system. To meet the requirements of 3.1.7, the contractor shall allocate the FCS safety budget among AWLS and other FCS. The specific probability of a hazard occurrence due to use of the landing system under an environment limit or operational restriction shall not increase the allocated probability by a factor of more than thirty. An alert height shall be established at an altitude such that, with all systems operative at the alert height, the probability of a hazard occurring during the landing is extremely remote, as defined in 6.6."

Requirement

3.1.7.1.1 Assessment of Average Risk of a Hazard. The average risk of a hazard due to use of the all-weather landing system shall be established considering:

a. The effect of each failure and combination of failures on system performance and the probability of their occurrence.

b. The effect of each relevant failure and combination of failures in systems operating concurrently with the all-weather landinf AFCS on aircraft performance and the probability of their occurrence.

c. The probability of the system not performing within the required levels as specified in 3.1.2.10 taken in conjunction with the probability that exceedance of those performance levels will result in a hazard.

Comparison

The C-5A contractual requirements included the requirement for hazard analyses to be conducted for the FCS. These analyses were qualitative and included an evaluation of the effect of component failures, the various modes of failure, primary and secondary results of the failure, corrective action, subsequent operation, and fail-safe substantiation. Particular attention was given to failure effects on interfacing systems and subsystems. These detailed hazard analyses were required for all potentially critical areas.

The quantitative probability of occurrence was not required for the FCS hazards that were analyzed.

Discussion

The definition of risk from a safety standpoint is obtained by multiplying the severity of a potential hazard, e.g.,

1 - Negligible
2 - Marginal
3 - Critical
4 - Catastrophic

by the probability of hazard occurrence. Therefore, the average risk referred to in paragraph 3.1.7.1.1 must be interpreted to be:

$$\frac{\Sigma \text{ (risks)}}{\text{total hazards}} = \text{average risk of a hazard}$$

or

$$\frac{\Sigma \begin{pmatrix}\text{severity}\\ \text{factor}\end{pmatrix}\begin{pmatrix}\text{probability}\\ \text{of occurrence}\end{pmatrix}}{\text{total hazards}} = \text{average risk of a hazard}$$

The term "average risk" has little significance to the analysis process as defined by MIL-STD-882, DID-DI-H-3278, or AFSC Design Handbook 1-6. Therefore, the deletion of the term average would provide a better understanding of the analysis process described by paragraph 3.1.7.1.1.

If "average" is deleted then the requirement would imply that for each hazard identified for normal AWLS operating conditions, a severity factor and probability of occurrence will be established. The product of these two values would be the risk value for each hazard.

Recommendation

It is recommended that the following changes be made to paragraph 3.1.7.1.1:

"3.1.7.1.1 Hazard Risk Assessment. The risk of a hazard incurred under normal operational conditions due to use of the all-weather landing system shall be established considering:

a. "leave as is"

b. "leave as is"

c. "leave as is"

Requirement

3.1.7.1.2 Assessment of Specific Risk. For each environmental limitation or operational restriction which limits the use of the all-weather landing system the specific risk shall be established. This evaluation shall comprise the average risk assessment, adjusted for a 1.0 probability of occurrence of environmental limits associated with the operational restriction.

Comparison

There was no C-5A quantitative requirement for assessment of specific risks for the AWLS function. A quanlitative analysis was required and accomplished. This was done through the hazard analysis process described in the discussion for paragraph 3.1.7.1.1.

Discussion

This paragraph imposes a requirement for analyzing the risk associated with the certain operation of the AWLS down to its environmental and operational limitations. Quantification of the risk termed "specific risk" is as follows:

Specific Risk = severity factor x probability of occurrence x 1.0

where:

severity factor is the hazard category, e.g.,;

1	= Negligible
2	= Marginal
3	= Critical
4	= Catastrophic

probability of occurrence = the quantitative value associated with the likelihood of hazard occurrence

1.0 = probability that the system will be operated at its environmental limitations or operationally restricted level

The specific risk value is interpreted to represent the quantitative assessment of operating the aircraft AWLS at or beyond its design or intended capability.

The term average risk assessment is used in the second sentence of the paragraph and confuses the issue of risk assessment. Elimination of the term "average" would clarify the intent of this paragraph.

Recommendation

It is recommended that the word "average" be deleted and "hazard" be inserted in its place in the second sentence of paragraph 3.1.7.1.2. The remainder of the paragraph should remain unchanged.

Requirement

3.1.8 Survivability. FCS Operational State IV or State V shall be provided as required by the procuring activity.

3.1.8.1 All Engines Out Control. For those aircraft which are dependent upon engine generation of flight control system power, supplementary means or power source shall be provided as necessary to supplement the control power available from the engine (s) where engines are unproven, airframe aerodynamics not established in flight, or windmilling power is insufficient to maintain operational State IV control capability anywhere in the aircraft operational envelope. Flight control system design (including power sources) shall be such that unintentional loss of any or all engine thrust shall not result in less than FCS Operational State IV including any necessary transition to emergency source(s) of power. Provision shall be made for inflight reversion to normal power wherein the transition shall not result in a worse FCS operational state.

Comparison

The C-5A was required to have adequate hydraulic power to maintain aircraft control in the event of the loss of power on all four engines. The engines would not windmill and consequently could not supply adequate hydraulic power in this way. In order to meet this requirement, a Ram Air Turbine (RAT) which drives a 32 gpm hydraulic pump is provided to pressurize hydraulic system No. 2 for this condition. Power is then supplied only to the flight controls powered by the No. 2 hydraulic system and the hydraulic driven emergency generator. The flight controls which can be actuated for use in maintaining operational State IV control capability are the inboard and outboard elevators, lower rudder, three flight spoiler panels per side, ailerons and pitch trim actuator nut drive. The hydraulic power distribution is shown in Figure No. 1(3.1.8.1).

The RAT provides sufficient power for a controlled descent in the clean configuration when all engines are lost with the landing gear in the stowed position, or with the gear down when all engines are lost in this configuration. The RAT provides capability for aircraft controllability using the emergency flight controls for as long a time as is required to restart the engines, to start an air startable APU, or for a time period of 15 to 20 minutes depending upon the rate of descent.

The RAT is operable at air speeds from 520 KTAS at 28,000 feet down to 135 KTAS at sea level and at altitudes from 45,000 feet down to sea level. The RAT is capable of delivering 32.0 GPM at 2,900 psi in less than 5.0 seconds after the RAT deployment switch is manually actuated, when deployed at air speeds in excess of 165 KCAS. At air speeds below 165 KCAS, the flow remains at 32.0 GPM, but the discharge pressure capability is reduced linearly down to 1470 psia at 135 KCAS.

The deployment system is designed for extension at air speeds of 135 KCAS up to 350 KCAS/0.825 Mach and retraction at air speeds below 200 KCAS. Extension time is 2.0 seconds and retraction time is 6.0 seconds. Once deployed, the RAT will not tear loose at aircraft limit speeds. An accumulator with pressurized

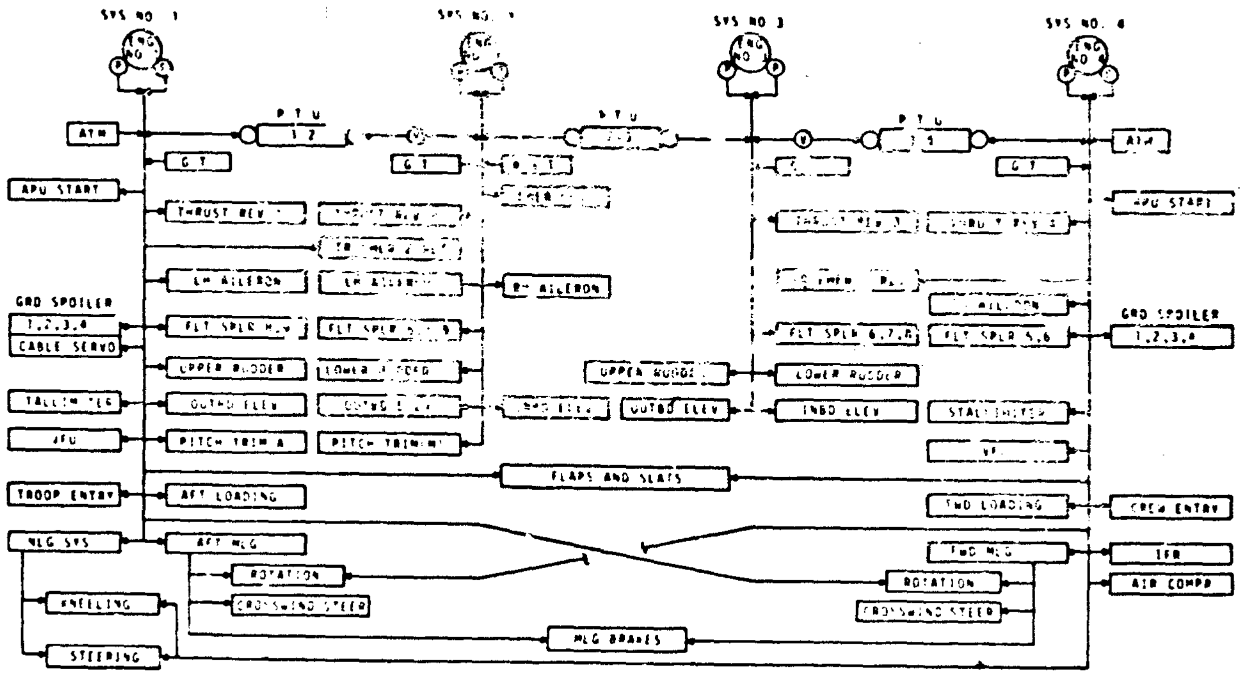


FIGURE 1 (3.1.8.1). HYDRAULIC SYSTEMS POWER AND DISTRIBUTION

fluid trapped behind a check valve supplies energy for deploying the RAT.

Electrical power for deploying the RAT is supplied by the battery bus. In flight, deployment of the RAT can be either automatic or manual. When the selector switch is in the automatic position, deployment of the RAT is automatic with the loss of three engines, i.e., two inboard engines and either outboard engine. The generator low frequency drop out circuits on the generator constant speed drive units are used as the sensing device for detecting the loss of three engines.

In conclusion, it can be stated that the C-5A is in compliance with the intent of this requirement.

Discussion

The extent to which the requirement addresses the subject is considered insufficient. The reason is that the need for the kind of standby system ordinarily used (Ram Air or Hydrazine powered) occurs infrequently. Unless an operational test of the system is required and performed checked periodically in flight, the reliability of the system cannot be demonstrated.

Recommendation

Add the following sentence to the requirement: "Requirements for periodic ground tests as determined by the contractor shall be incorporated into the airplane maintenance handbook and requirements for periodic checkout in flight shall be provided when necessary."

Requirement

3.1.9 Invulnerability. Degradation in flight control system operation due to variations in natural environments, adverse events of nature, induced environments, onboard failure of other systems, maintenance error, flight crew error or enemy actions shall be within the following limits.

3.1.9.1 Invulnerability to Natural Environments. Flight control systems shall be designed to withstand the full range of natural environmental extremes established for the particular vehicle or system without permanent degradation of performance below FCS Operational State I, or temporary degradation below FCS Operational State II. Reductions below State I shall be experienced only at adverse environmental extremes not normally encountered and shall be transient in nature only; and, the function shall be recovered as soon as the aircraft has passed through the adverse environment. System components and clearances with structure and other components shall be adequate to preclude binding or jamming, instability, or out of specification operation of any portion of the system due to possible combinations of temperature effects, ice formations, loads, deflections, including structural deflections, and buildup of manufacturing tolerances.

3.1.9.2 Invulnerability to Lightning Strikes and Static Atmospheric Electricity. Flight control system shall maintain State II capability or better when subjected to electric field and lightning discharges as specified in MIL-B-5087 and in AFSC Design Handbook DH 1-5, except that a temporary, recoverable, more extensive loss of performance to State III is allowable in the event of a direct lightning strike.

3.1.9.3 Invulnerability to Induced Environments. Flight control systems shall withstand the full range of worst case induced temperatures and temperature shock, acceleration, vibration, noise and shock, induced pressures, explosive and corrosive atmospheres, electromagnetic interference (EMI), and nuclear radiation, including electromagnetic pulse, projected in missions for the particular aircraft, without permanent degradation or loss of capability to maintain FCS Operational State II capability. These induced environments within structural and crew survival limits shall not result in temporary degradation during the exposure to the environment below FCS Operational State IV capability. The FCS shall meet the requirements of MIL-A-8892, MIL-A-8893, and the applicable requirements of MIL-E-6051 and MIL-STD-461.

3.1.9.4 Invulnerability to Onboard Failures of Other Systems and Equipment. The FCS shall meet its failure state/reliability budget, as allocated within the weapon system, for self-generated failure (within the FCS) and for those FCS failures induced by failures of other interfacing systems within the weapons system (3.1.6, 3.1.7). In addition, the FCS design shall comply with the following:

a. Essential and flight phase essential flight control systems shall retain FCS capability at Operational State III (minimum safe) or better after sustaining the following failures:

(1) Failure of the critical engine in a two-engine aircraft

(2) Failure of the two most critical engines in aircraft having three or more propulsive engines

(3) Failure of any single equipment item or structural member which, in itself, does not cause degradation below State III. This includes any plausible single failure of any onboard electrical or electronic equipment in any subsystem of the aircraft.

b. Flight control systems, including the associated structure and power supplies on MIL-F-8785 Class III aircraft, shall be designed so that the probability of losing the capability of maintaining FCS operation to no less than State IV as a result of an engine or other rotor burst is extremely remote (6.6).

c. Flight control systems, including the associated structure and power supplies on MIL-F-8785 Class I, II, and IV aircraft, shall be designed so that the probability of degrading FCS operation below State V as a result of an engine or other rotor burst is extremely remote (6.6).

3.1.9.5 Invulnerability to maintenance error. Flight control systems shall be designed so that it is physically impossible to install or connect any component item improperly without one or more overt modifications of the equipment or the aircraft. Provisions for adjusting the flight control system on the aircraft, except during initial buildup, major overhaul, or rigging during major maintenance activities, shall be minimized. All line replaceable units (LRU's) shall be designed to permit making internal adjustments only on the bench. The system shall require only a minimum of rerigging following replacement of LRU's. In addition, all control linkages and other flight control mechanisms shall be designed to resist jamming from inadvertent entry of maintenance tools or other material.

3.1.9.6 Invulnerability to pilot and flight crew inaction and error. Flight control systems shall be designed to minimize the possibility of any flight crew member controlling or adjusting system equipment to a condition state which could degrade FCS operation.

a. Protection against improper position and sequencing of controls - Wherever practical, cockpit controls, other than stick or wheel and rudder pedals, shall be equipped with positive action gates to prevent inadvertent positioning which can compromise safe operation of the aircraft. Positive interlocks to prevent hazardous operation or sequencing of switches shall be provided.

b. Protection against inflight engagement of control surface locks.

c. Pilot reaction to failure - Flight control systems shall be designed to that the normal pilot reaction to cues provided by probable failure conditions is instinctively correct.

d. Warning requirements:

(1) Warning information shall be provided to alert the crew to unsafe system operating conditions. Systems, controls, and associated monitoring and warning means shall be designed to preclude crew errors that create additional hazards.

(2) A clearly distinguishable warning shall be provided to the pilot under all expected flight conditions for any failure in a redundant or monitored flight control system which could result in an unsafe condition if the pilot were not aware of the failure.

3.1.9.7 Invulnerability to enemy action. Essential and flight phase essential flight control systems, including associated structure and power supplies, on all aircraft designed for combat operations shall withstand at least one direct encounter from the threat defined by the procuring activity without degradation below Operational State III.

Comparison

3.1.9 Invulnerability. The following validation comparisons discuss the requirements which limited the degradation in FCS operation under the influence of the various defined environmental conditions. Generally, only a temporary performance degradation (such as reduced response and variations in pilot feel forces) was encountered when the aircraft FCS was subjected to the adverse conditions of the environment.

3.1.9.1 Invulnerability to Natural Environments. The C-5A FCS was designed to comply with the natural environment requirements of Contract End Item (CEI) specification CP 40002-1A. The air vehicle was designed to be capable of executing the required design mission in all conditions of weather and climate in any area of the world using the MIL-STD-210 values for temperature, humidity, rain, snow, dust, and atmospheric pressures. The wind gust criteria for the FCS was met as noted in the validation discussion for Paragraph 3.1.11 structural integrity. The criteria for natural environment defined in AFSCM 80-1, Volume I, Part A, Chapter 8, was used in the definition of specific conditions which included those discussed above in addition to solar radiation, lightning, fog ice-fog, dew, hail, icing, sleet, frost, salt-spary, sand, clouds, and fungus. The aircraft design met the environmental requirements by the appropriate selections of materials, components, and systems. The aircraft used environmental control systems for functions related to control of personnel and equipment environment, such as the aircraft pressurization and temperature control system. Other external systems were used

such as rain removal (windshield), ice protection (for airfoils and equipment) and lighting systems. The aircraft FCS components and clearances with structure and other components was adequate to preclude binding or jamming, instability, or any other system degradation due to combinations of temperature effects, ice formations, loads, deflections, and manufacturing tolerances. The aircraft was designed to withstand the required range of natural environmental extremes without permanent degradation of the FCS performance below operational State I or temporary degradation (of a transitory nature) below operational State II. FCS critical subsystems and functional components were subjected to environmental testing in accordance with MIL-STD-810 and the aircraft was subjected to extreme environmental climatic tests, including the MIL-T-5289 test requirement.

3.1.9.2 Invulnerability to Lightning Strikes and Static Atmospheric Electricity. The C-5A aircraft was designed to provide protection against a lightning strike such that the lightning discharge current can be carried between any two points on the aircraft without the risk of damage to the FCS. The guidelines of MIL-B-5087 were followed to provide lightning protection and electrical bonding. Electrical bonding and/or lightning arrestors were provided for stroke guidance for any electrically isolated conducting objects or structure in the aircraft. The significance of lightning strikes on the C-5A FCS is reduced to some degree by the fact that dual mechanical control systems are the primary FCS modes whereas the more vulnerable electrical augmentation systems serve a lesser role. Precipitation static (electricity) was reduced by the use of static dischargers.

3.1.9.3 Invulnerability to Induced Environments. The C-5A FCS was designed to comply with the induced environment requirements of CEI specification CP 40002-1A and the guidelines of AFSCM 80-1, Volume I, Part A, Chapter 8. The design requirements covered the full range of induced environments resulting from the operational envelope of the aircraft operating within the worldwide natural environment extremes as defined in MIL-STD-210 and discussed above. Induced environments considered such areas as high temperature, temperature shock, vibration, mechanical shock, noise, acceleration, explosive or corrosive vapors, nuclear radiation, sound, electromagnetic interference, and exhaust gases. The FCS elements were designed to operate satisfactorily without loosening, malfunction, or failure in the induced vibration, mechanical shock, and acceleration environment which may emanate from the flight envelope or engine or equipment--acoustic energy or mechanical vibrations. Direct and induced temperature changes versus time and their effects on the aircraft, FCS, FCS elements and FCS power supplies which could arise within the flight envelope both without and during simultaneous operation of adjacent equipment or systems were considered. The FCS was not adversely affected by the electromagnetic interference (EMI) limits which were designed to MIL-STD-826 or the electrical power transients which were limited as defined by MIL-STD-704.

3.1.9.4 Invulnerability to Onboard Failures of Other Systems and Equipment. The C-5A aircraft and FCS met the CEI CP 40002-1A design goals for mission accomplishment reliability and flight safety. These requirement goals precluded (within the defined limits) self-generated failures within the FCS and FCS failures which may have been induced by failure of other interfacing systems and equipment. The reliability goals were to achieve operation of the aircraft such that 90 percent of all dispatched sorties reach their destination without a major subsystem failure, an additional 8 percent of the sorties may be subjected to failures which do not abort the mission and up to 2 percent of the sorties may experience failures which would allow aborting at departure or landing short of the intended destination. The flight safety goals stated, "The air vehicle shall be designed to the highest standard of fabrication, function, and operation and shall incorporate those safety features necessary to insure that failure of the air vehicle or of its components will not create a hazardous condition by reason of its mode of failure or by the direct effect of such failure on the air vehicle, related equipment or personnel."

The above conditions are contingent upon reasonable aircraft condition, proper maintenance, and qualified crew. Additional elaboration is made on these areas in the validation discussions for Paragraphs 3.1.6, "Mission Accomplishment Reliability," and 3.1.7, " Quantitative Flight Safety."

The C-5A FCS has no single function which is essential or flight phase essential as noted in the validation discussion for Paragraph 1.2.3, "FCS Criticality Classification." Multiple FCS failures are required to degrade the FCS below operational State III. This includes failure of the two most critical engines or hydraulic systems or failure of any single equipment item or structural member which, in itself, does not cause degradation below operational State III. The aircraft can maintain operational State IV after the loss of all four engines by the deployment of a Ram Air Turbine (RAT) as noted in the validation discussion for Paragraph 3.1.8.1, "All Engine Out Control."

3.1.9.5 Invulnerability to Maintenance Error. The C-5A FCS was designed to achieve CEI design goals for qualitative and quantative maintainability characteristics such that the planned mission could be accomplished with a minimum (guaranteed) number of manhours, elapsed time, personnel skills, ground equipment and technical data. This helped to develop a FCS which was simple and less vulnerable to maintenance error.

The FCS design required that it be physically impossible to install or connect any element or component incorrectly without extensive modification. An example is the cable attachment shown in Figure No. 1 (3.2.3.2.4.3) where the cable turnbuckles are staggered such that cross connection of the cables is impossible. All adjustment devices have mechanical locking provisions, such as the turnbuckle locking clip shown in Figure No. 2 (3.2.3.2.4.3).

The FCS, for the design goal of fouling prevention, was designed to have the transmitting elements protected and was configured to resist jamming by foreign objects as noted in the validation discussion for Paragraph 3.2.3.1.3.

The number of rigging positions and operations were kept to a minimum with easy accessibility as noted in the validation discussion for Paragraph 3.2.3.1.4. See Paragraph 3.1.10.3 for additional validation discussion for ready accessibility and serviceability of FCS components which reduced the vulnerability of the FCS to maintenance errors. All the FCS LRU components were designed to require internal adjustments to be made on the bench. All the "bench adjustment" areas on the LRU's were sealed and stamped by the cognizant inspection authorities. Replacement of LRU's only requires a minimum of rerigging of the FCS.

3.1.9.6 Invulnerability to Pilot and Flight Crew Inaction and Error. The C-5A FCS was designed to provide indications to minimize the possibility of any flight crew member maneuvering system equipment to a condition state which could degrade FCS operation.

As a protection against improper positioning and sequencing of FCS action, the cockpit controls are equipped with positive action indication. The pitch, roll and yaw axes controls have neutral position centering and controller feel force systems which provide increasing pilot action forces with increasing controller and control surface movement. The roll and yaw axis FCS have the same force gradient for all speeds, but the pitch axis has an increasing force gradient as a function of increasing aircraft speed.

The rudder surface deflection capability is proportionately limited as a function of increasing airspeed by a rudder limiter in the input system. The pilot can remove the rudder limiter stops by actuation of a "rudder limiter" switch on the overhead instrument panel.

The flight spoiler (roll control) panel deployment authority is limited automatically by retraction of the wing flaps. Deployment of the wing flaps automatically restores full authority to and slightly up rigs the flight spoiler panels. The secondary controls, wing flaps, ground spoilers, and horizontal stabilizer trim have pilot actuated control handles which have position detents and markings. In addition, these functions, as well as all trim functions, have instrument indicators which indicate the control surface position to the pilot.

The ground spoiler control handle is locked in the closed position in flight. The spoiler handle lock is normally removed by the sequence of operations of the throttles in or aft of the minimum cruise position, wheel spin-up, and aircraft touchdown. A throttle interconnect automatically moves the ground spoiler handle to the closed position when the throttles are advanced to take-off power.

The ground spoiler handle lock-out can be overridden by the pilot. The pilot has visual indication of the ground spoiler handle lock-out position.

The surface locks cannot be engaged in flight since this function is assured by check valves within the hydraulic servo actuators when the hydraulic systems are de-energized and the input control valves are held in a stationary position. Even when the hydraulic systems are de-energized in flight, pilot movement of the primary controllers will move the main control valve allowing hydraulic fluid within the actuators to bypass and let the surface float. The pilots' normal and emergency flight procedures outlined in the operational flight manuals contain descriptions of cues and indications provided to the crew by probable failure conditions and define the corrective action required to control the aircraft. Failure indications may be provided by indicator lights and instruments, changes in the aerodynamic response of the aircraft, or changes to the controller response and/or feel forces.

Redundant systems such as the primary FCS stability augmentation have a continuous on-line monitor and failure detection system which automatically switches channels and gives indication to the pilot following a failure mode. All failure modes were examined to determine that recognizable failure indications would be forthcoming so that corrective action could be taken within the allowed recognition/response time by the trained pilot.

3.1.9.7 Invulnerability to Enemy Action. The C-5A FCS was designed with a level of system redundancy and separation to permit at least one direct encounter of enemy action damage equivalent to the failure modes defined in the validation discussions for Paragraphs 1.2.2 and 1.2.3. Since there are no single FCS functions which are essential or flight phase essential multiple failures would be required to degrade the FCS below operational State III as noted in Table No. 4 (1.2.3) and Table No. 5 (1.2.3). This would include failure of the two most critical engines or hydraulic systems. The aircraft can maintain operational State IV after the loss of all four engines by deployment of the RAT.

Discussion

These are all valid requirements which have been satisfied by the C-5A FCS design and can be demonstrated. These requirements should be specified for all future transport type aircraft.

Recommendation

Retain the requirements as stated.

Requirement

3.1.10 Maintenance Provisions. FCS design and installation shall permit normally available maintenance personnel to safely and easily perform required maintenance under all anticipated environmental conditions. Means shall be provided to facilitate the accomplishment of all required maintenance functions including: operational checkouts, system malfunction detection, fault isolation to the LRU (line replaceable unit) level, LRU removal and replacement, inspection, overhaul, servicing, and testing.

Comparison

The C-5A flight control systems were designed with performance, reliability, maintainability, and simplicity as primary objectives. Provisions for all the maintenance functions included in the requirement were incorporated in the design of the systems. The C-5A flight control systems comply with the intent of this requirement.

Discussion

There are no additional items recommended for inclusion in this requirement.

Recommendation

Accept "as is."

Requirement

3.1.10.1 Operational Checkout Provisions. Flight control systems shall be designed with provisions for operation on the ground, without operating the main engines, to verify system operation and freedom from failure to the maximum extent practical. They shall be designed to operate with the power generation subsystems supplied by standard Air Force ground carts, as specified by the procuring activity or by self-contained power supplies.

Comparison

The C-5A flight control systems are capable of being operated and completely checked out on the ground without having to operate the main engines. There are two ways electrical and hydraulic power can be supplied to the systems. One way is to operate the APU. The APU provides 115 volts AC and 28 volts DC electrical power as well as hydraulic power directly to systems No. 1 and No. 4. The Air Turbine Motor (ATM) driven pumps provide 50 GPM to each system. This flow rate is sufficient to check out each control system, one at a time.

Another way to check out the control systems is to supply external electrical and hydraulic power to the aircraft using the Type A/M32A-60 generator set and the MJ-2 hydraulic power unit. Their power sources are standard Air Force AGE and are available at all the Air Force facilities where C-5A's are based.

Figure No. 1 (3.1.8.1) shows the hydraulic schematic indicating the tie in of the APU and ATM units. Figure No. 1 (3.1.10.1) is the hydraulic control panel showing the ATM switching functions.

Discussion

This is a valid requirement which has been satisfied by the C-5A design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

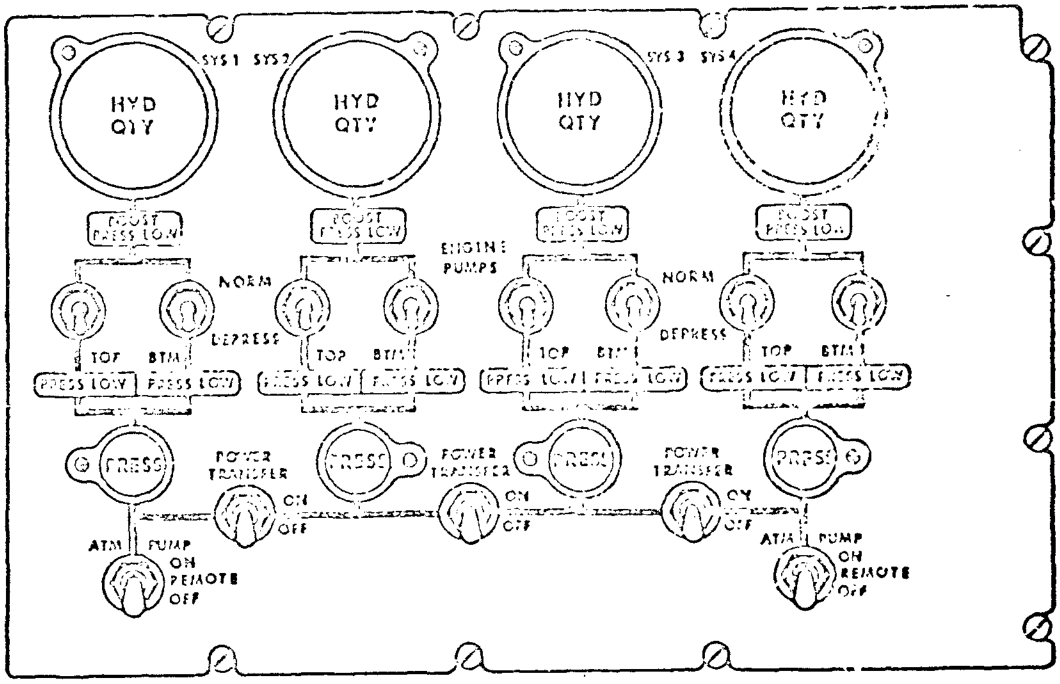


FIGURE 1 (3.1.10.1). FLIGHT ENGINEER'S HYDRAULIC PANEL

Requirement

3.1.10.2 Malfunction Detection and Fault Isolation Provisions. Means providing a high probability for detecting failures and monitoring critical performance conditions as required to isolate faults to the LRU level shall be incorporated in all flight control electrical and electronic systems required to perform essential and flight phase essential functions. These means may include cockpit instrumentation and built-in test equipment. For the mechanical and fluid power portions of the flight control system, provisions for the use of portable test equipment may also be incorporated as required to meet the maintenance support and operational concept of the particular weapon system.

Comparison

The C-5A was designed with malfunction detection and isolation to a LRU as a basic design requirement. The isolation of a failure to a LRU utilized the built-in tests (BIT), flight instrument warning flags, annunciators, system automatic monitoring, and the Malfunction and Detection Analysis and Recording System (MADAR).

Throughout the C-5A design requirements the integration of malfunction isolation into the design was prevalent as shown in this excerpt from the CEI specification on Flight Controls, CP 40002-6B:

> "3.3.1 General Design Features. The Flight Control Subsystem shall be simple, direct and foolproof as possible with respect to design, operation, inspection and maintenance. Emphasis shall be placed on simplicity of mechanization, ease of maintenance, and minimum of dependence on Aerospace Ground Equipment...."

The AFCS systems listed below were originally designed as essential and flight phase essential, but are not now considered as such, meet the intent of the requirements of this paragraph.

- Angle of Attack System
- Stallimiter System
- Roll, Yaw and Pitch Stability Augmentation
- Autoland System

Techniques and indicators are available in the C-5A which may be used to isolate functional failures in the mechanical and fluid power portions of the flight control system. Most of the failure modes for the MFCS and AFCS and the related indications and corrective actions are covered in detail in the FCS failure effects analysis in Lockheed Report LG1US42-2-1.

The MFCS maintainability functions serve as a malfunction detection means; i.e., during ground checkout of the systems. Portable test equipment and ground electrical and hydraulic power carts are used to support these functions.

Discussion

It is felt that the requirement for a "high probability for detecting failure" is not conclusive and the requirement stated in Paragraph 3.1.3.9.1, first sentence, should be used here but be deleted from Paragraph 3.1.3.9.1.

Recommendation

Revise the requirement as follows:

> Change first sentence from: "Means providing a high probability for detecting failures...." to "Means providing a 90% probability for detecting failure...."

Requirement

3.1.10.2.1 Use of Cockpit Instrumentation. Where acceptable procedures result or are provided, cockpit instrumentation may be used for malfunction detection and fault isolation where it provides readily understandable condition indication either alone or in coordination with built-in test equipment, or with portable test equipment (for nonelectrical and nonelectronic components).

Comparison

The C-5A MFCS utilizes cockpit instrumentation for malfunction detection and fault isolation. The AFCS utilizes self-test along with the cockpit indicators to isolate to a failure. No portable test equipment is used. The C-5A meets the intent of this requirement.

Discussion

The C-5A experience indicates that there is a level of BIT or self-test beyond which it becomes very costly or impractical to isolate to the faulty LRU. Therefore, the use of some type of carry-on break-out panels or test equipment should be allowed in these remote cases.

The last sentence of this requirement references "portable test equipment (for nonelectrical and nonelectronic components)." This statement indicates that portable test equipment can be used only for MFCS, yet requirement 3.1.10.2.2 allows the use of portable test equipment under specific conditions.

Recommendation

Revise the requirement as follows:

> Delete "(for nonelectrical or nonelectronic components)" from end of paragraph.

Requirement

3.1.10.2.2 Provisions for Checkout with Portable Test Equipment. Where the use of built-in test equipment would cause excessive penalties and where the use of portable test equipment is compatible with the maintenance support concept, provisions shall be made to permit the use of generally available and commonly used portable test equipment. Components which require peculiar, special, or new items of test equipment shall be avoided.

Comparison

The C-5A meets the intent of this paragraph since the FCS checkout is performed without the use of portable test equipment.

Discussion

The last sentence of this requirement does not allow any improvement in the state-of-the-art in FCS beyond that which would exist at the time this specification is imposed. In addition, the last sentence will not allow a design if the aircraft mission or construction requires new equipment which then requires new test equipment.

The use of generally available and commonly used portable test equipment should be emphasized, but the design should not be constrained completely due to a test equipment requirement.

Recommendation

Revise the requirement as follows:

> Change last sentence to read, "Components which require peculiar, special, or new items of test equipment should be avoided unless dictated by aircraft design, mission requirements, or state-of-the-art improvements."

Requirement

3.1.10.3 Accessibility and Serviceability. Components shall be designed, installed, located, and provided with access so that inspection, rigging, removal, repair, replacement, and lubrication can be readily accomplished. Suitable provisions for rigging pins, or the equivalent, shall be made to facilitate correct rigging of the control system.

Comparison

The C-5A FCS was designed to the CEI maintainability requirements which were contractually guaranteed and had to be demonstrated. The requirements specified qualitative and quantative maintenance characteristics to accomplish the air vehicle mission with a minimum of maintenance manhours, elapsed time, personnel skill, ground equipment, and technical data. Service and access for maintenance of the aircraft were achieved by having the design of the air vehicle, equipment, and ground support equipment consider the requirements of maintainability, reliability, self-sufficiency, and human factors. The detail accessibility and serviceability considerations included the following:

- The design of mechanical/hydraulic/electrical control elements were such that removal and replacement could be accomplished with a minimum disturbance of the rigging.

- The use of tools and ground support equipment made maximum use of the stock inventory equipment and minimized the need for special maintenance equipment.

- All maintenance areas were accessible to personnel wearing artic clothing and with a minimum removal of access panels or plates, covers, and adjacent components or equipment. Access panels were equipped with quick disconnect fasteners.

Internal access areas such as the empennage used an integral access ladder as shown in Figure No. 1 (3.2.3.2.4).

Figure No. 2 (3.2.3.2.4) shows the arrangement of access panels in the lower wing for servicing the roll control system and represents a typical access arrangement.

All equipment or components which were physically interchangeable also had to be functionally interchangeable. Where practical, FCS equipment and LRU's had to be capable of being handled by one man without the use of special ground handling equipment. Where practical, test points and power receptacles were located and identified in a central location, for a particular subsystem.

The MFCS was designed to accommodate easy servicing and rigging and to have a minimum number of adjustments. The MFCS used a minimum number of rig pins. The rig pins were made accessible and identified by streamers in critical

applications. To assure a cleared and functioning system a rigid functional checkout was required of any system subjected to a rigging maintenance action.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.1.10.4 Maintenance Personnel Safety Provisions. Systems and components shall be designed to preclude injury of personnel during the course of all maintenance operations including testing. Where positive protection cannot be provided, precautionary warnings or information shall be affixed in the aircraft and to the equipment to indicate the hazard, and appropriate warnings shall be included in the application maintenance instructions. Safety pins, jacks, locks, or other devices intended to prevent actuation shall be readily accessible and shall be highly visible from the ground or include streamers which are. All such streamers shall be of a type which cannot be blown out of sight such as up into a cavity in the aircraft.

Comparison

The C-5A FCS was designed to the personnel safety and ground safety requirements of CEI CP 40002-1A. The ground safety requirements were met by having ground operations, maintenance procedures and maintenance equipment designed to minimize human error or failure of equipment, injury to personnel, and damage to the air vehicle. The air vehicle and FCS design allowed the maintenance operations to be performed without significant hazard to the service personnel. Where hazardous maintenance operations could have resulted from incorrectly following procedures, the maintenance directives emphasized caution and explicit procedures; where practical, precautionary warnings were affixed to the FCS components.

The placement of maintenance access areas were designed to place personnel adjacent to the equipment or components in a manner which kept them as clear as practical from the moving elements. Personnel restraint harnesses were mandatory for all maintenance in high areas and which were not accessible to work stands such as on top of the wing or horizontal stabilizer.

Maintenance procedures for the FCS makes use of rig pins or safety locks to prevent inadvertent movement or actuation of other systems. During maintenance operation of the FCS, close coordination between all personnel--such as personnel in the cockpit operating the controls, external observers, and personnel adjusting or checking the equipment--must be constantly maintained using electrical/electronic communication equipment. Other maintenance operations which could possibly be in conflict or cause a hazard to the one being performed, were prohibited. All personnel were instructed to keep clear of control surfaces and moving elements of the FCS during actuation of the systems.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and can be readily demonstrated. This requirement should be specified for all future transport typ aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.1.11 Structural Integrity

3.1.11.1 Strength. The overall flight control system shall be designed to meet the applicable load, strength, and deformation requirements of MIL-A-8860, MIL-A-8861, MIL-A-8865, MIL-S-8698, and MIL-STD-1530. The components of the systems shall be designed in accordance with the strength requirements of MIL-A-8860, MIL-C-6021, MIL-F-7190, MIL-A-21180, MIL-A-22771, MIL-F-83142, MIL-HDBK-5, and MIL-HDBK-17.

3.1.11.1.1 Damage Tolerance. Those structural elements of the flight control system that are essential to safety of flight (to control essential and flight phase essential functions) shall meet the damage tolerance requirements of MIL-A-83444.

3.1.11.1.2 Load Capability of Dual-Load-Path Elements. The load path remaining after a single failure in dual-load-path elements shall meet the following requirements:

a. Where the failure is not evident by visual inspection or by obvious changes in control characteristics, the remaining path shall be capable of sustaining a fatigue spectrum loading based on one overhaul period. The time interval corresponding to an overhaul period shall be established by the contractor. The remaining path shall also withstand, as ultimate load, loading equal to 1.5 times the limit loads specified in MIL-A-8865, or 1.5 times the load from an alternate source, such as a powered actuation system or loads resulting from aerodynamic or other forces, if such load is greater.

b. Where the single failure is obvious, the remaining load path shall be capable of withstanding, as ultimate load, loading equal to 1.15 times limit loads specified in MIL-A-8865, or 1.15 times the load from an alternate source, such as a powered actuation system or loads resulting from aerodynamic or other forces, if such load is greater.

Comparison

The C-5A FCS was designed to meet the structural integrity requirements specified in the Contract End Item (CEI) documents CP 40002, Volumes 1, 2 and 6. The applicable military specifications included MIL-A-8860, MIL-A-8861, and MIL-A-8865 for general aircraft system strength and rigidity requirements. General materials properties were selected in accordance with MIL-Handbooks-5 and -17. Where applicable aluminum castings were designed to MIL-C-6021 and aluminum forgings were designed to MIL-F-7190 and MIL-A-22771. The uses of any new state-of-the-art materials were subject to the most current specification in effect at that time and to the approval of the customer. The FCS design application assured that no elements of the system were subjected to operation, either intermittently or continuously, at loads greater than those for which the part had been designed. The FCS elements were designed to withstand the design limit loads and thermal effects without detrimental

deformation. The design limit load was the maximum load normally required for aircraft operation. The FCS elements were designed not to yield at the specified limit load. The ultimate load was obtained by multiplying the design limit load by 1.5. All margins of safety in the stress analysis were shown to be no less than zero. All FCS elements were subjected to static and fatigue load testing to verify compliance with the CEI requirements. Additional detail discussion on the FCS structural integrity is contained in the validations for Paragraphs 3.1.11.3, 3.1.12, 3.2.3.2.1, 3.2.3.2.2, 3.2.6.1.1, 3.2.6.1.2, 3.2.6.4, and 3.2.6.7.1.1.

The philosophy to achieve a fail safe design included the use of multiple load paths and actuation systems. A typical example is shown in Figure No. 3 (3.2.3.1.1) for pitch axis control system. The FCS configuration design goals included providing for failure detection during routine service inspections and incorporated design practices to minimize crack initiation and to preclude their propogation.

The load capability of dual load-path elements provided for each element of the input system to be good for a limit load of 75 percent of two pilots. The dual actuation systems were designed to provide the required limit load margin for the remaining system following a single failure.

The damage tolerance requirements were essentially covered by the CEI criteria for the FCS redundancy and were approved by the customer.

Discussion

This is a good requirement which has been essentially satisfied by the C-5A control system design and can be readily demonstrated. MIL-A-21180 was not applicable to the C-5A since no high strength aluminum alloy castings were used. The C-5A flight control systems were designed to requirements which met the intent of MIL-A-83444 for aircraft damage tolerance requirements and MIL-STD-1530 for aircraft structural integrity program requirements. Therefore, the impact of full compliance would be minor. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.1.11.2 Stiffness. The stiffness of flight control systems shall be sufficient to provide satisfactory operation and to enable the aircraft to meet the stability, control, and flutter requirements as defined in the applicable portions of MIL-F-8785, MIL-A-8870, MIL-F-83300 and MIL-A-8865. Normal structural deflections shall not cause undesirable control system inputs or outputs.

Comparison

The C-5A FCS design meets the requirements of MIL-A-8870 and MIL-F-8785 which required the aircraft to be free of any flutter, buzz, divergence, and other dynamic aeroelastic, aerothermo-elastic, and aeroservoelastic instabilities of the aircraft and its relevant components at all speeds up to 1.15 V_L for all design ranges of altitudes, thermal conditions, and maneuvers where losses in rigidity could occur. Compliance was demonstrated by flight test and other test data, in addition to analytical data for the increase of 15 percent in equivalent airspeed for all relevant design points on the permissible flight envelope.

The aircraft fail-safe stability for speeds up to V_L was demonstrated to be free from flutter, divergence, or other aeroelastic instabilities following any single probable malfunction of the main FCS, including the augmentation system. The primary FCS control surfaces were mass balanced to preclude flutter in the event of a loss of both hydraulic systems, except the outboard elevators which utilized a triplex hydraulic actuator system from three separate hydraulic systems. In addition the inboard elevator surfaces use hydraulic dampers to guard against low frequency oscillation in the event of a multiple failure mode.

MIL-F-83300 for V-STOL aircraft was not applicable to the C-5A design. The applicable FCS sections of MIL-A-8865 were met.

Discussion

This is a valid requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. The "Users Guide" (AFFDL-TR-74-116) discussion on this requirement is excellent and is a good example of the type of coverage which should be provided for user information. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.1.11.3 Durability. Flight control systems shall be designed to meet the durability requirements of MIL-A-8866 and equal to that of the airframe primary structure considering the total number of ground and flight load cycles expected during the specified design service life and design usage of the aircraft from all commands; e.g., from the MFCS, AFCS, servo feedback and from load inputs. The requirements of MIL-A-8892 regarding vibrations and MIL-A-8893 regarding sonic fatigue also apply to the FCS.

Comparison

The C-5A air vehicle fatigue useful life design goal was 20 years or 30,000 hours of service life of which 6 percent was low level flight capability and 12,000 landings of which 5 percent were to be on support area airfields, which was specified in CEI CP 40002-1A. The applicable requirements of MIL-A-8866 Airplane Strength and Rigidity Reliability requirements, repeated loads and fatigue, were to the extent as specified in CEI CP 40002, Vol. 2. The FCS sub-systems and component elements were designed and endurance tested to have a fatigue life equal to or better than the overall air vehicle useful life. The specific fatigue requirements were derived from mission profile analysis, by contractural specifications or a combination of these two. Refer to the validation discussions of Paragraphs 3.1.12, Wear Life, and 3.2.6.4.3, Actuating Cylinders, for additional detail fatigue life requirements for the FCS.

The C-5A FCS was designed to the environmental vibrations requirements of MIL-A-8870.

Discussion

This is a good requirement which has been satisfied by the C-5A to the extent specified in the CEI specification and it can be readily demonstrated. Although the vibration and sonic fatigue specifications MIL-A-8892 and MIL-A-8893 were not applicable at the time of the C-5A contract the intents of these requirements were imposed either by other specifications and other analyses evaluations. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Additional Data

The following is a typical servoactuator endurance life requirement table. This example is for a rudder servoactuator. The load/cycle curve reflects the hinge moment versus surface deflection curve which was followed for

the surface deflection indicated. The input commands may be provided by a combination of mechanical and electrical (E-H Valve) inputs. Endurance Life - The servoactuator shall be capable of performing the number of loaded and unloaded stroke cycles, applied through the input system specified in the table below:

Load/Cycle Curve	No. of Cycles	Rudder Surface Displacement from Faired Position	
"E"	2,000	35° left	35° right(100%)
"F"	8,000	11.0° left	10.5° right
"G"	280,000	5.5° left	5.3° right
"H"	410,000	1.1° left	1.0° right
"J"	3,300,000	0.2° left	0.2° right
"K"	20,000(unloaded)		
"L"	290,000(unloaded)	3.5° left	3.5° right
"M"	700,000(unloaded)	0.7° left	0.7° right

Requirement

3.1.12 Wear Life. Mechanical elements of the FCS shall be designed to have wear life equal to the wear life specified for the overall aircraft. Parts subject to wear, such as hydraulic seals, bearings, control cables, sensors and hydraulic actuator barrels, may be replaced or their wearing surfaces renewed after they exceed their useful life. However, all replacements shall be within the FCS wear out-replacement budget established for the overall weapon system. Electronic and other nonmechanical LRU's shall remain economically repairable and shall meet reliability requirements throughout the specified airframe lifetime.

Comparison

The useful life design goal for the C-5A aircraft was 20 years or 30,000 hours of service life as defined in the CEI specification SP 40002-1A. Mechanical elements of the FCS were designed to have wear life equal to the overall aircraft either by mission profile analysis or by contractual specification. The wear life requirements paralleled the fatigue life requirement cycles as required by Paragraph 3.1.11.3 and meet or exceed requirements specified for the primary aircraft structure. The primary FCS servo actuators were required to meet a minimum of 5,000,000 endurance life cycles of various load and stroke combinations of anticipated amplitudes, frequencies and surface actuator loads which were related to the cyclic distribution defined in MIL-C-5503C. All other FCS met endurance life cycles, the criteria specified in CEI CP 40002-6B, which were compatible with the air vehicle life requirements. The design goal of having the component parts wear life equal to the aircraft life was reflected in the maintenance philosophy which permitted component replacement on condition rather than on schedule. An example of this is the FCS hydraulic servo actuator seal replacement. The MIL-SPEC permitted seal replacement every 500,000 cycles, the detail component specifications contained a design goal for seal life of at least 5,000,000 cycles endurance life. Many of the C-5A servo actuators, designed by the Bertea Corporation of California, completed extended endurance testing of 10,000,000 cycles or twice the required life. All FCS LRU's were designed to meet overhaul cost parameters (which were specified as a percentage of the original installation cost) and accessibility and maintainability goals. These were designated to assure incorporation of economically repairable units and to meet the reliability requirements for aircraft life. See comparison section under Paragraph 3.1.11.3 for additional discussion of fatigue life.

Discussion

This is a good requirement which has been satisfied by the C-5A FCS, can be readily demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2 Subsystem and Component Design Requirements

3.2.1 Pilot Controls and Displays. Wherever a FCS control, display or annunciator is interfaced with redundant flight control channels, mechanical and electrical separation and isolation shall be provided to make the probability of common mode failures at least extremely remote. FCS controls and displays shall be designed in accordance with MIL-STD-1472.

Comparison

The C-5A FCS has redundant flight control channels in the AFCS and in the dual mechanical load paths for the pitch and roll control systems. Mechanical system redundancy and separation is provided to preclude what might be considered "common mode failure." This is discussed in detail in the validation discussion for Paragraphs 3.2.3.1.1, Control Element Routing, and 3.2.3.1.2, System Separation, Protection, and Clearance. Examples of the dual mechanical systems are shown on Figures Nos. 1 (3.2.3.1.1), 2 (3.2.3.1.1) and 3 (3.2.3.1.1).

The AFCS incorporates redundant flight control channels, each capable of performing the given functions and including failure detection monitoring which accomplishes display annunciation and automatic switching. Mechanical and electrical separation and isolation are provided by placement of switches, displays, actuation components, and wire routing. Figure No. 1 (3.2.1), Pitch Augmentation Subsystem Schematic Diagram, represents the typical AFCS stability augmentation system. The redundant circuits are shown for channels 1 and 2 for the LVDT and E-H valve control circuits. The forward overhead panel display, shown on Figure No. 3 (3.2.1.1.7) shows the dual switch arrangement which controls the hydraulic power distribution for the inboard elevator servo actuators. Figure No. 2 (3.2.1) is a portion of the AFCS interface block diagram which shows the ties between the FCS controls, displays, and the annunciator. Figure No. 3 (3.2.1) indicates the pitch augmentation fault logic. Figure No. 4 (3.2.1) indicates the pitch augmentation fault logic. Figure No. 4 (3.2.1), AFCS Control Panel and Annunciator Panel, shows the controls and indicators arrangement.

The C-5A AFCS controls and displays were designed to the human engineering design criteria specified in Contract End Item CP 40002-6B which used criteria and guidelines from MIL-H-27894, MIL-STD-803, and AFSCM80-1, 80-3, and 80-6. The C-5A AFCS was designed to achieve operational reliability and to maintain separation of independent functions with respect to operation from common power supplies and for maintenance requirements. The AFCS utilizes separate engagement of axes except for the autopilot roll and yaw axes which utilize a common switch. The system was designed to preclude undesirable control transients when switching from one functional mode to another or when disengaging the system from a steady state condition unless otherwise dictated by operational requirements.

Redundancy circuitry and mechanization was employed to ensure that no single

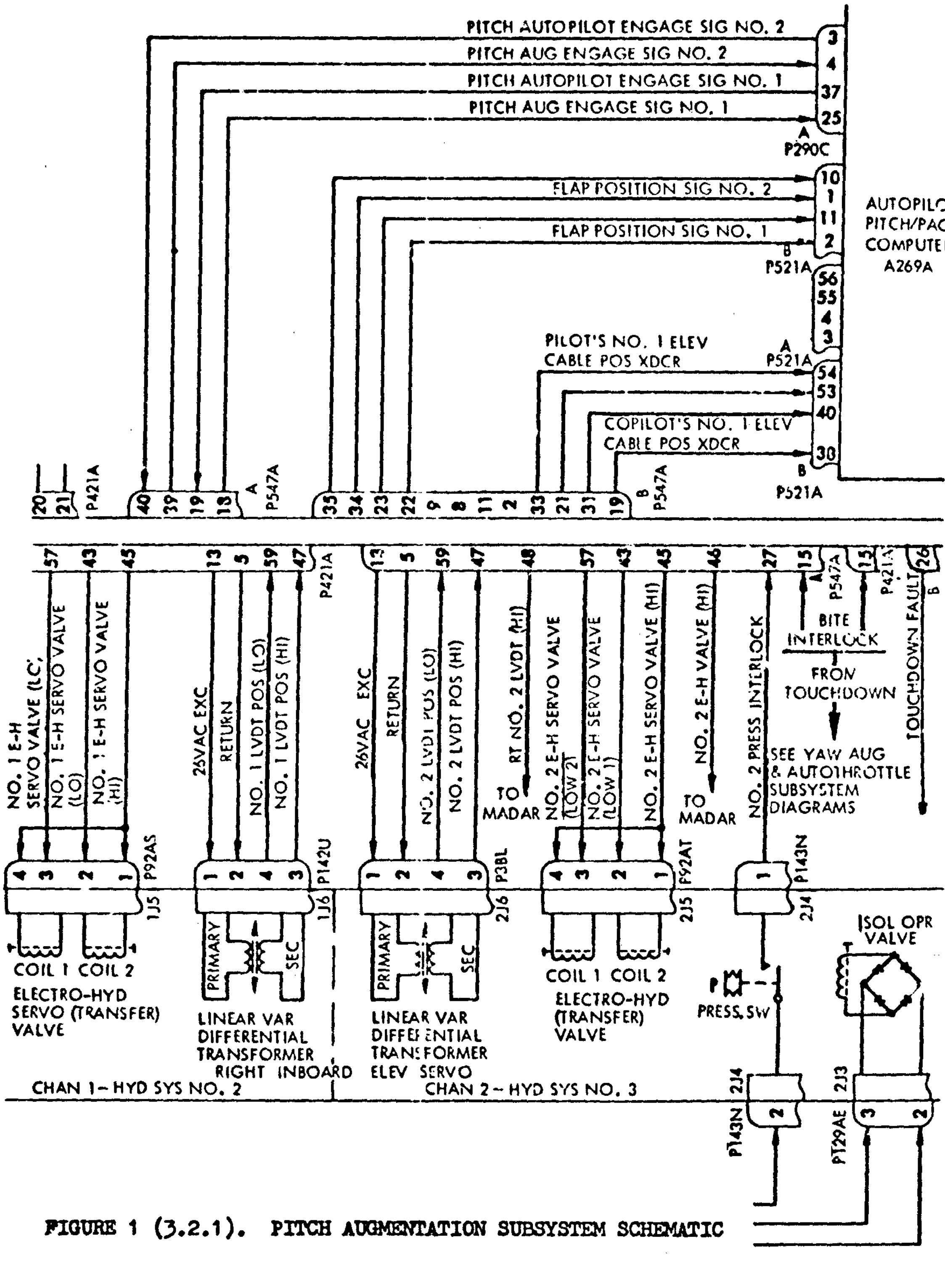


FIGURE 1 (3.2.1). PITCH AUGMENTATION SUBSYSTEM SCHEMATIC

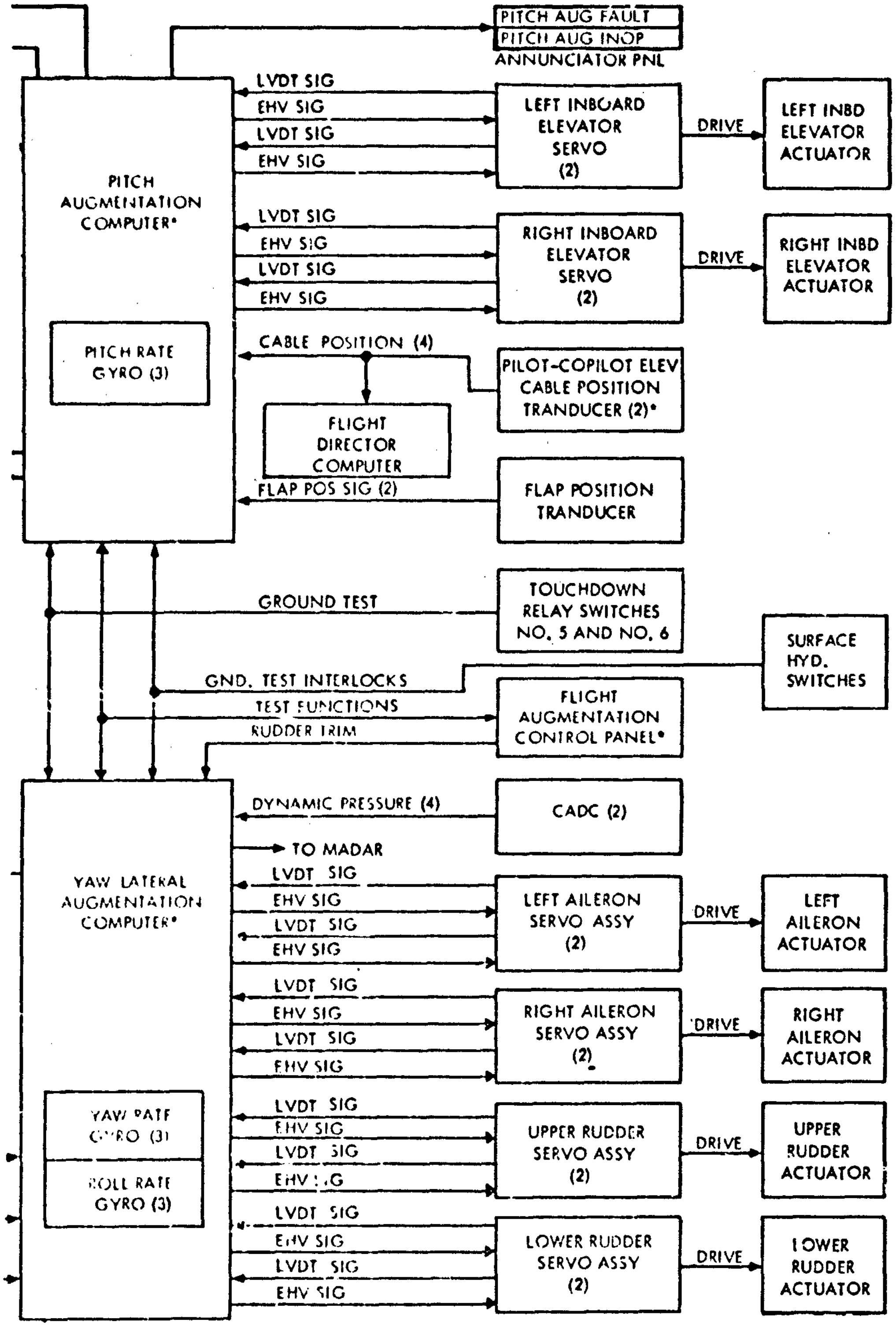


FIGURE NO. 2(3.2.1) AFCS INTERFACE BLOCK DIAGRAM

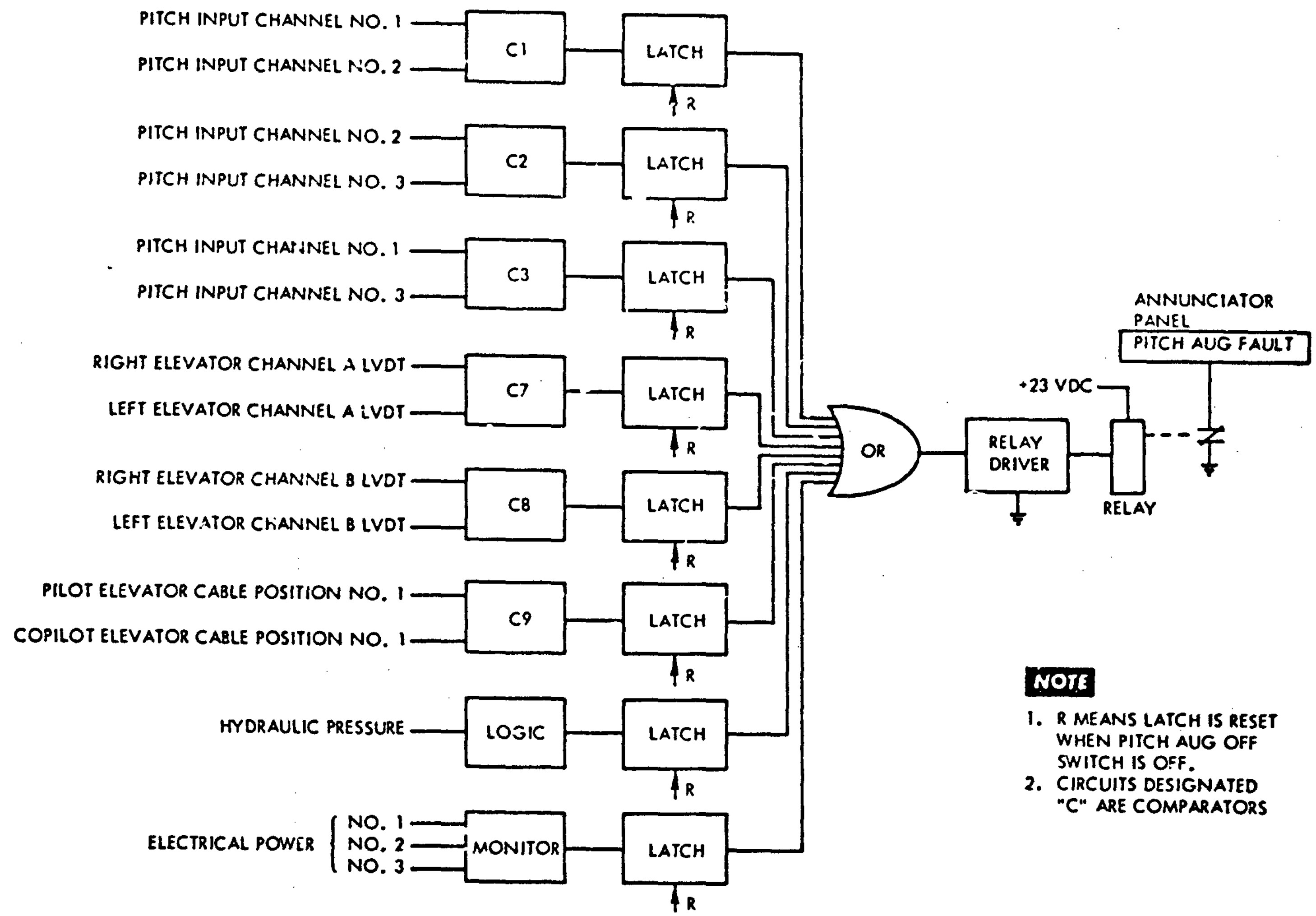
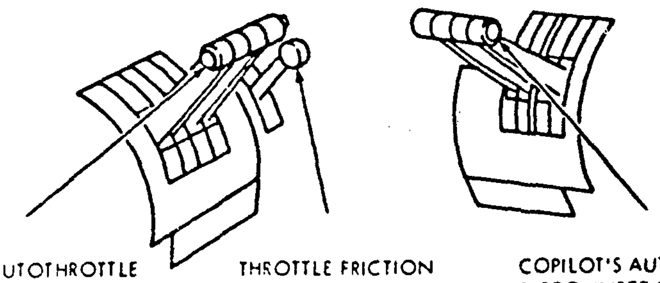
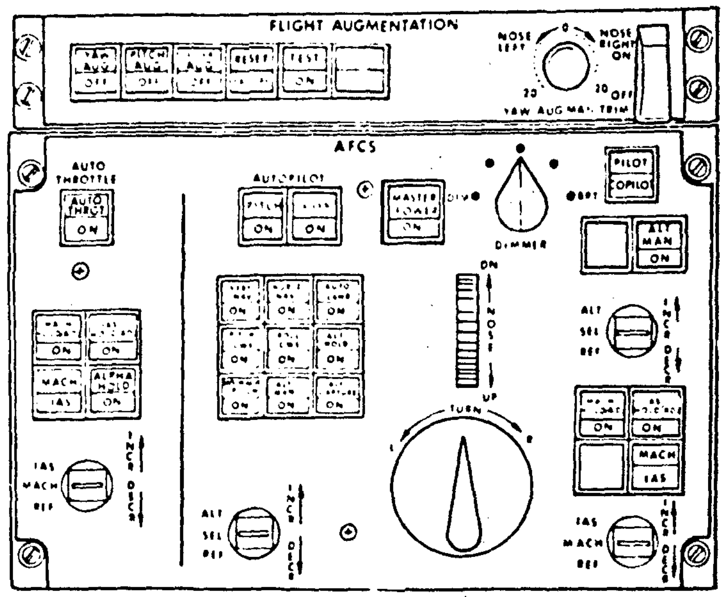


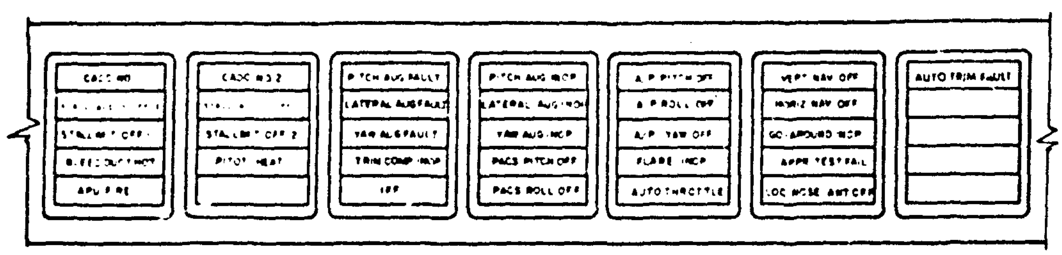
FIGURE 3 (3.2.1). PITCH AUGMENTATION FAULT LOGIC DIAGRAM



FLIGHT AUGMENTATION PANEL



AFCS CONTROL PANEL



PORTION OF ANNUNCIATOR PANEL

FIGURE NO. 4(3.2.1) AFCS CONTROLS AND INDICATORS

malfunction could cause catastrophic damage to the aircraft.

Discussion

The last sentence of this requirement is too stringent and should be changed to allow the designer to utilize all design parameters in addition to the MIL-STD-1472 human engineering criteria and guideline for an integrated system. The C-5A used MIL-STD-803 in the design of controls and displays for criteria and guidelines. MIL-STD-803 was superseded by MIL-STD-1472 which is a general specification on human engineering aspects to the design and does not specifically dictate basic design of controls or displays. It is felt that the C-5A meets the intent of this requirement, but a revision should be made to the last sentence to clarify the intent. This will prevent a great number of formal deviations which would otherwise be unnecessary since cockpit arrangements and displays are always a compromise between many requirements and subject to customer concurrence with mockups.

Recommendation

Revise the last sentence of the requirement as follows:

> "FCS controls and displays shall be designed using the criteria contained in MIL-STD-1472 as a guideline and shall be subject to approval of the procuring agency.

Requirement

3.2.1.1 Pilot Controls for CTOL Aircraft. Pilot's cockpit controls for conventional take-off and landing (CTOL) aircraft shall be designed and located in accordance with AFSC Design Handbook DH 2-2, DN 2A1, Aircrew Controls; DN 2A5, Flight Controls; and the following subparagraphs. Strict adherence to the prescribed location and maximum range of motion of these controls is required.

Comparison

The C-5A was designed to the requirement below: Ref. CP 40002-6B.

Pilots' Controls - The pilots' cockpit controls shall be designed and located using MIL-STD-203, MS33574, and MS33576 as guides.

Control Wheels - The control wheels shall be of the W type, 14 to 16 inches in diameter. They shall be constructed of a light-weight, non-hygroscopic, non-slippery, non-stick black material with a low heat conductivity. The forward face of the portions gripped by the hands shall have corrugations to fit the fingers and provide a good finger-type grip surface.

Rudder Pedals - Rudder pedal size, shape, travel and adjustment mechanism shall be designed using MS33574, MS33576, MIL-B-8584, and MIL-STD-203 as guides. The foot pedals shall be interconnected to insure positive movement of each pedal in both directions. The pedal motion shall be straight line relative to the seat reference point. The pedal travel shall be in a plane inclined approximately 7 degrees, 52 minutes from the horizontal plane with the pedal reference point located at least 5.0 inches above the heel rest line.

For the actual C-5A arrangement, see Figure No. 1 (3.2.1.1) and Figure No. 2 (3.2.1.1).

Discussion

The C-5A Flight Station Basic Dimensions in Figure No. 1 (3.2.1.1) show several differences when compared to the present requirement.

The C-5A requirement was shown as a guide only whereas MIL-F-9490D states strict adherence. The differences are tabulated below:

Item	C-5A	MIL-F-9490D
1. Seat Angle	6° to 26° from Vert.	13.5° to 33.5° from Vert. MIL-S-25073A
2. Angle of Rudder Pedal Movement	7°46' Down	Horizontal

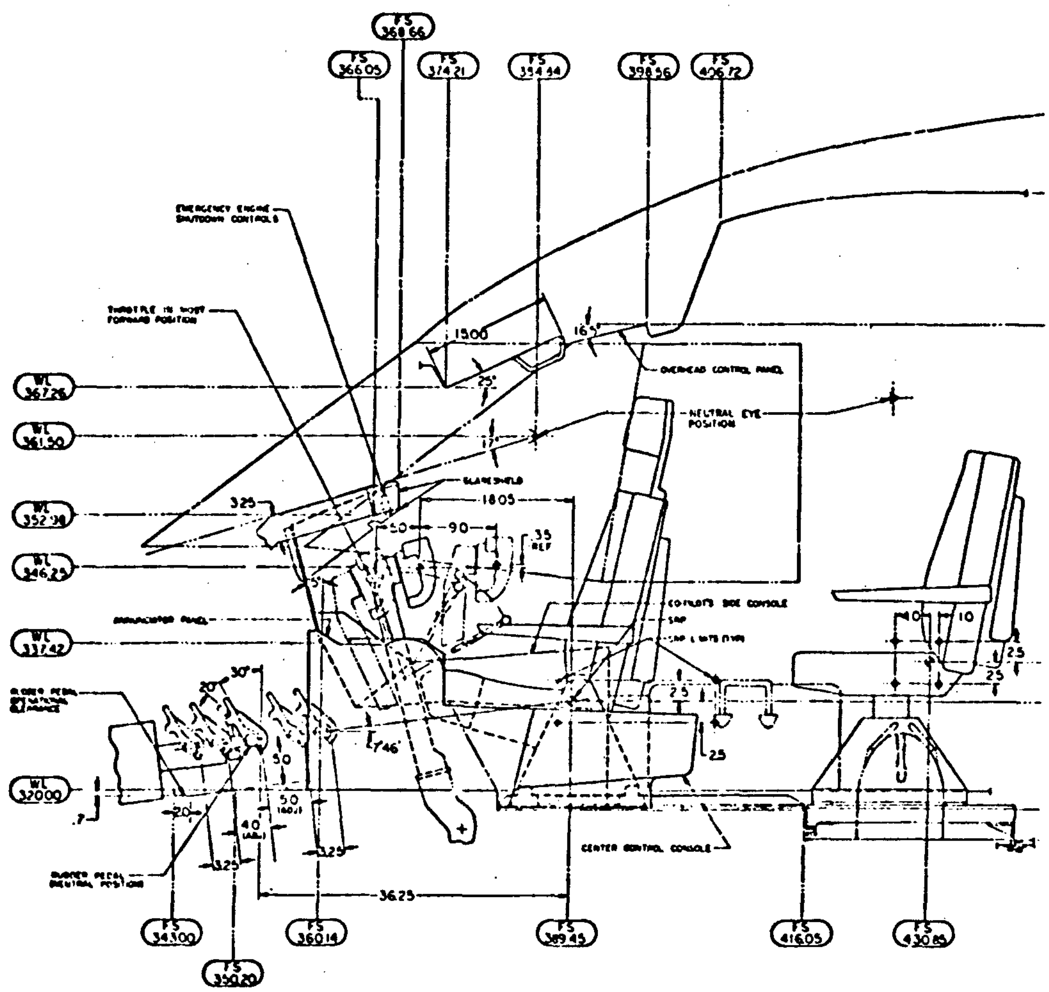


FIGURE 1 (3.2.1.1). C-5A FLIGHT STATION ELEVATION

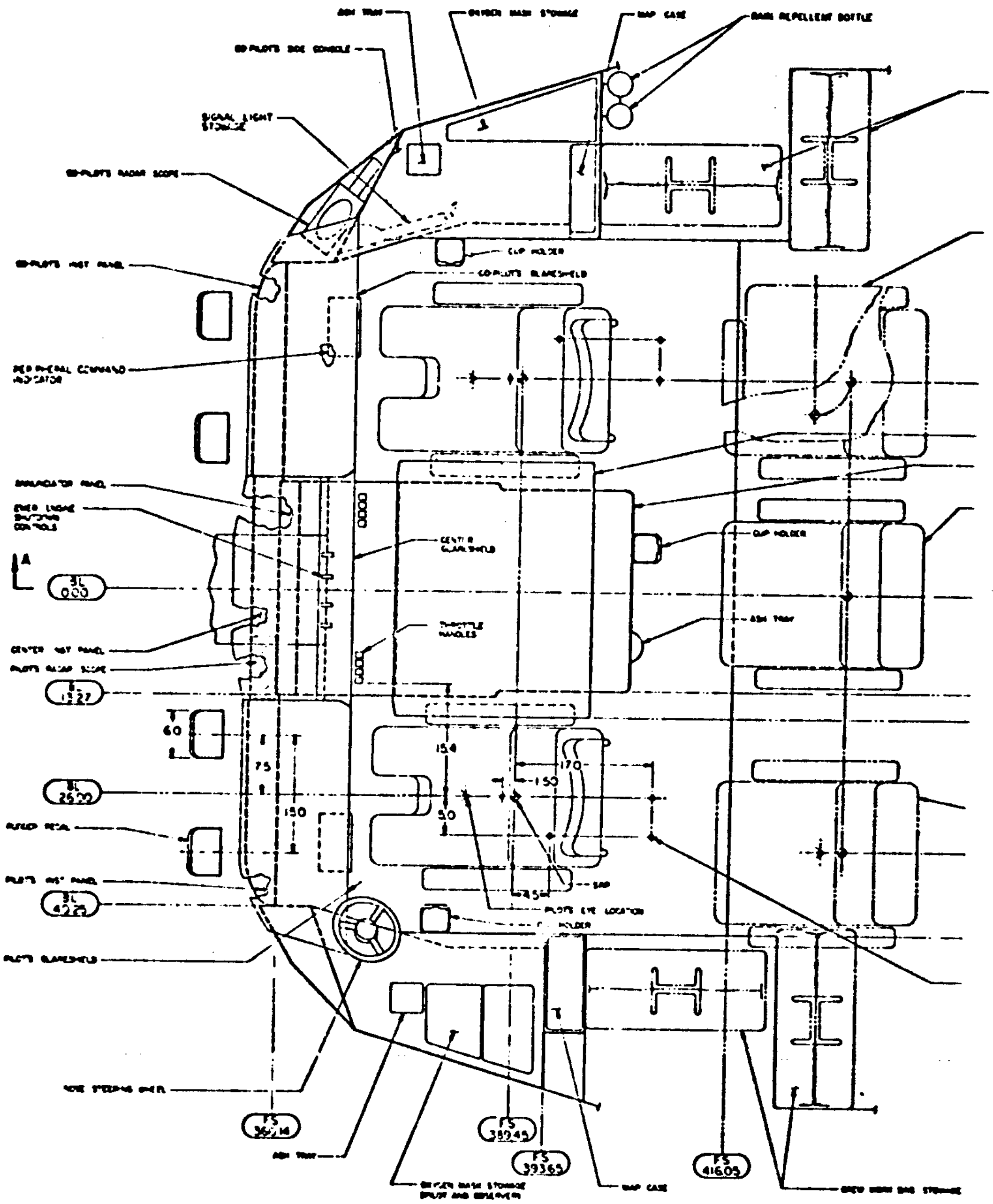


FIGURE 2 (3.2.1.1). C-5A FLIGHT STATION PLAN

The C-5A seat adjustment angle (Item 1) was originally set to the specification requirement (13.5° to 33.5°) per MIL-S-25073. During the flight test program, the test pilot could not get a satisfactory view of the runway without leaning forward because of his elevated location above the runway. The seat adjustment angle was changed so that the pilot could adjust his seat ack to provide the pilot with a proper view of the runway. The rudder pedal travel on the C-5A was set at an angle of 7°46' down (Item 2) because of the limited space available.

The seat angle and vision clearance requirements of MIL-F-9490D are not compatible with the C-5A. The elevation of the pilot affects his visibility. Although there is similarity in the provisions and arrangements of cockpit controllers between CTOL transport aircraft, each is different. These differences generally result from compromises with other requirements such as outside visibility and limitations such as cockpit space and are evaluated in a mockup subject to customer approval and flight validation. The requirement should be a goal or guideline, but not one to be strictly adhered to when such adherence may result in a less desirable overall product.

<u>Recommendation</u>

Delete the last sentence of the requirement.

<u>Requirement</u>

3.2.1.1.1 <u>Additional Requirements for Control Sticks</u>. Not applicable.

Requirement

3.2.1.1.2 Additional Requirement for Rudder Pedals. Rudder pedals shall be interconnected to insure positive movement of each pedal in both directions.

Comparison

C-5A rudder pedals are mechanically and rigidly interconnected at each pilot's station so that movement of one pedal fore or aft will cause the other pedal to move in the opposite direction. In addition the pilot's and copilot's pedals are interconnected so that both sets of pedals work in a coordinated fashion throughout the travel.

Discussion

The requirement is valid and did not create any unusual problems in meeting it in the C-5A. It is valid for future transport aircraft.

Recommendation

Accept the requirement as is.

Requirement

3.2.1.1.3 Alternate or Unconventional Controls. If pilot's controls other than the conventional center located sticks, W-type wheels, rudder pedals, trim controls and indicators, wing incidence control, wing sweep control, landing flap control and indicator, speedbrake control, and automatic flight control panels specified in AFSC Design Handbook DH 2-2, DN 2A5, are utilized, demonstration of their adequacy and suitability is required prior to installation in an aircraft.

Comparison

The C-5A FCS used conventional controls therefore this requirement did not apply.

Discussion

Lockheed has considered alternate or unconventional controllers in many different design studies and proposals and agrees that extensive development and demonstration of their adequacy and suitability should be required prior to their installation in an aircraft. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.1.1.4 Variable Geometry Cockpit Controls. Not applicable.

Requirement

3.2.1.1.5 Trim Switches. Electrical trim system switches of the five-position, center-off, toggle type shall be in accordance with MIL-S-9419. Control stick grips in accordance with MIL-G-25561 shall already have the trim switches, conforming to MIL-S-9419, installed. Three-position trim switches shall be approved switches similar or equivalent to the MIL-S-9419 switches.

Comparison

Trim switches are utilized in the C-5A rudder, aileron and pitch trim control systems. The pitch trim switches on the control wheels conform to MIL-S-6743 with Notice 2. The aileron, rudder and alternate pitch trim switches on the center console conform to MIL-S-3950 with Supplement 1. Since none of these switches, shown on Figures No. 1 (3.2.1.1.5), and No. 2 (3.2.1.1.5) conform to MIL-G-25561 or MIL-S-9419, the C-5A trim system switches do not conform to the requirements of Paragraph 3.2.1.1.5.

Discussion

Lockheed believes that this requirement is too restrictive and that it should be revised to include additional military specifications covering trim switches not included in MIL-G-25561 and MIL-S-9419. This position is based on the requirement that no single failure may cause a pitch trim runaway as implied by Paragraph 3.2.1.1.6 of MIL-F-9490D. It is not possible to meet this requirement with a single MIL-S-9419 switch because two signal paths - demanding two separate switches - are necessary to meet this requirement.

Recommendation

It is recommended that the last sentence of Paragraph 3.2.1.1.5 be revised as follows:

> "Three-position trim switches shall be approved switches similar or equivalent to the MIL-S-9419, MIL-S-3950 or MIL-S-6743 switches."

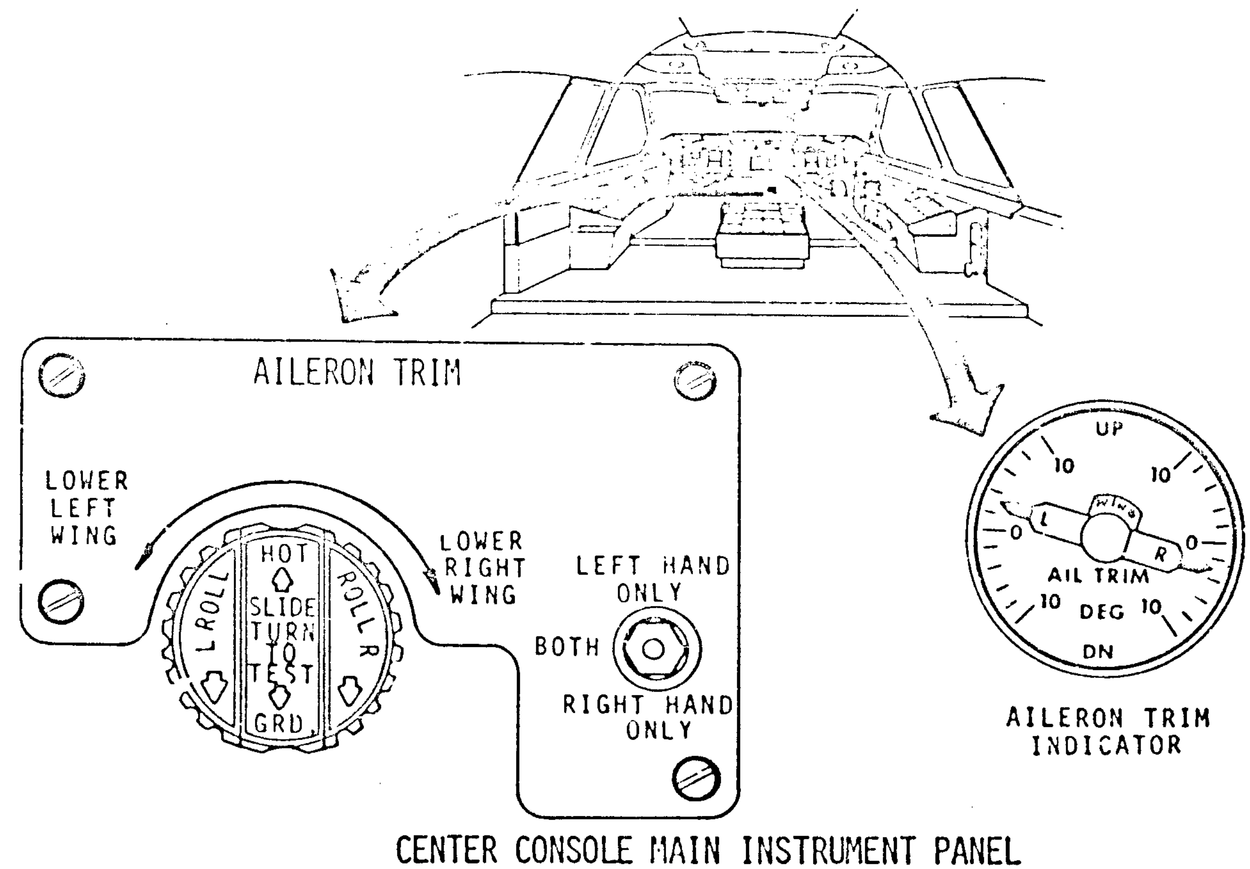


FIGURE 1 (3.2.1.1.5). AILERON TRIM CONTROL SYSTEM

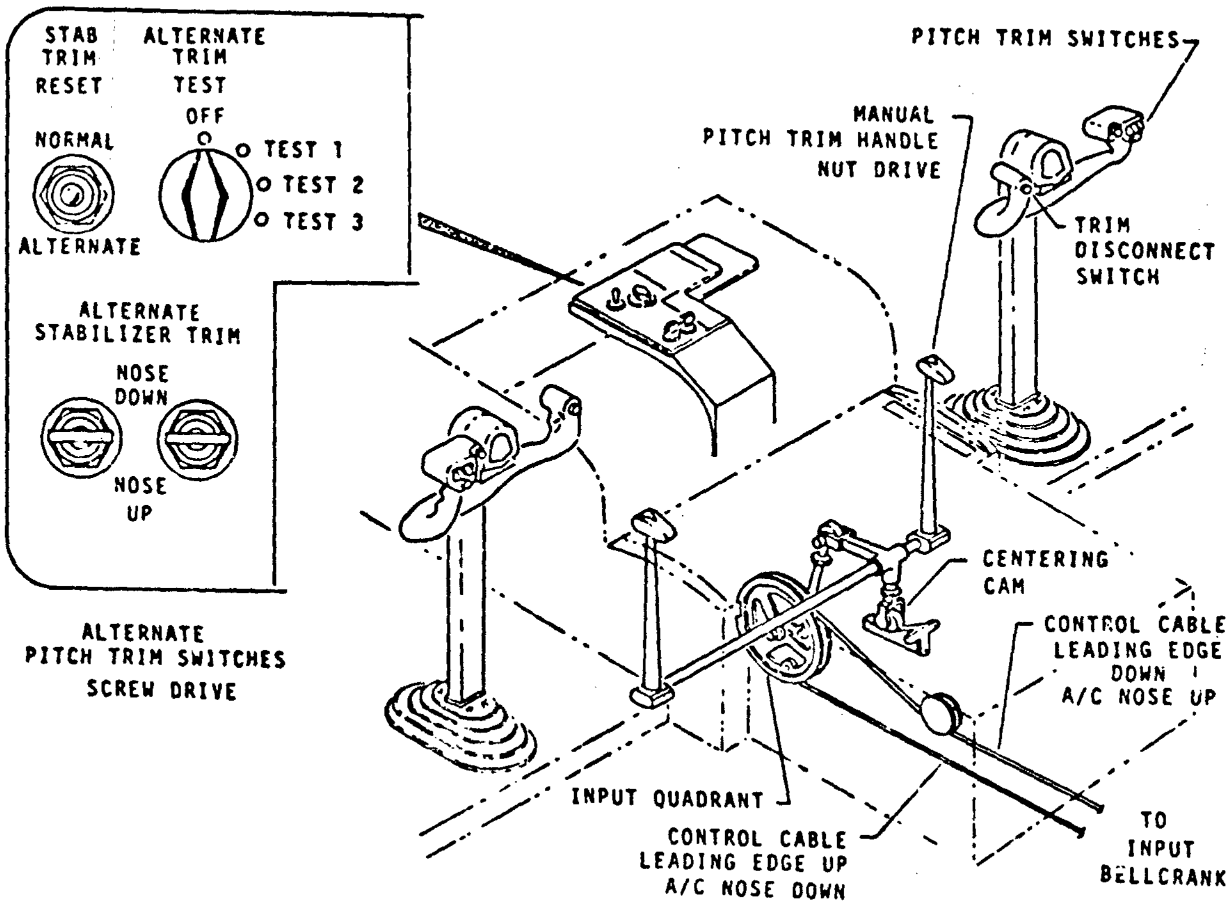


FIGURE 2 (3.2.1.1.5). PITCH TRIM CONTROL SYSTEM

Requirement

3.2.1.1.6 Two-Speed Trim Actuator. Two-speed trim actuator systems shall be designed to preclude runaway or inadvertent operation in the high-speed trim mode.

Comparison

The only two-speed trim actuator used on the C-5A is the pitch trim actuator shown with its input control modes in Figure No. 1 (3.1.3.5). This actuator is used to trim the aircraft about the pitch axis by movement of the horizontal stabilizer and is independent of the primary pitch control system (elevators). The pitch trim actuator, which provides a dual structural load path between the horizontal stabilizer and the vertical stabilizer, is an irreversible linear screwjack actuator. The screw is the upper section of the actuator and is powered by a hydraulic motor through a no-back device and reduction gear train. The nut is the lower section and is also powered by a hydraulic motor through a no-back and a reduction gear train. The hydraulic sources for the screw drive and nut drive are completely independent of each other.

A high degree of safety and reliability is provided since two signals are required from the input system before the actuator can operate. The screw drive unit is commanded by either of the two following modes:

1. Alternate Mode - The pilot or copilot can operate the pitch trim system in this mode by depressing two switches on the center console in the desired direction. The dual switch operation sends an electrical signal to the screw drive hydraulic manifold shut-off valve and directional control valve. These valves port fluid to the hydraulic motor which drives the screw through the connecting gear train.

2. Autopilot Mode - When the pilot selects this mode of operation, the autopilot will send commands to the screw drive hydraulic manifold and will operate the same valves as the alternate system.

In either of the above modes of operation, the screw will continue to drive until the command signal is stopped or until the horizontal stabilizer operates a limit switch. The limit switch interrupts the command signal and no further trim is possible in that selected direction through the screw drive mechanism.

The nut drive unit is commanded by either of two modes.

1. Normal Mode - The pilot or copilot can operate the pitch trim system in this mode by operating simultaneously dual switches, spring loaded to the center off position, on the outboard grip of either control wheel. The dual switch operation sends an electrical signal to the nut drive hydraulic manifold shut-off

valve and directional control valve. These valves port fluid to the hydraulic motor which drives the nut through a connecting gear train. The nut will continue to drive until the wheel switches are released or until the horizontal stabilizer operates a limit switch. The limit switch interrupts the command signal and no further trim is possible in the selected direction through the wheel switches.

2. Manual Mode - The pilot or copilot can operate the pitch trim system in the manual mode by moving a lever on the side of the center console (there is a lever provided on each side of the console). At the top of lever there is a spring loaded trigger type switch. Depressing the handle switch accomplishes two things. It sends an electrical signal to the nut drive hydraulic manifold shut-off valve which then ports fluid to the directional control valve. It also sends a 28 VDC signal to the autopilot which disconnects the unit while the manual trim system is operating. Moving the lever forward or aft moves the directional control valve by means of a cable and mechanical linkage. Fluid is then ported to the nut drive motor which drives the nut in the selected direction through the gear train. The nut will continue to drive until either the trigger switch is released, the trim handle is brought back to neutral or the actuator reaches the mechanical stops. This system is not affected by the horizontal stabilizer limit switches while the aircraft is airborne.

The trim rate for the screw drive mechanism is .33 inches per second for all flap configurations. The trim rate for the nut drive mechanism is .67 inches per second with the flaps up and 1.12 inches per second with the flaps down and/or with the aerial refueling door open. This trim rate is the change in the pitch trim actuator length with time between upper and lower mounting points.

Pitch Trim Disconnect Circuit. Located on the inboard grip of the pilot and copilot control wheels is a pitch trim disconnect switch. This switch when operated provides a ground for two disconnect relays. The first relay disconnects the 28 VDC power source from the normal trim control system. The second relay sends a disconnect signal to the autopilot trim system and disconnects the 28 VDC power source from the alternate trim system. Therefore, when the pilot or copilot operates the trim disconnect switch, all pitch trim modes, except the manual mode, are disconnected. These trim modes will remain disconnected until a reset switch, located above the alternate trim mode switch, is operated.

The reset switch has three positions. The center position is neutral. The up position operates the relay connecting the normal trim system. The down

position operates the relay connecting the alternate and autopilot systems.

In the design of the pitch trim system, consideration has been given to possible malfunctions and their effect on the controllability of the aircraft. The most dangerous condition is that of a runaway trim actuator which can be deactivated by the trim disconnect system. The system has been designed to insure that no single mechanical or electrical malfunction will cause a runaway actuator. In addition, the fact that two switches must be actuated in the normal and alternate modes and actuation of a switch together with a lever in the manual mode precludes inadvertent operation. Therefore, the pitch trim system meets the intent of the requirement.

Discussion

The requirement is considered too lenient as written since it does not require inadvertent operation to be prevented in any trim mode. In addition, it does not require the use of a trim disconnect switch to disconnect all pitch trim modes except for the manual or emergency trim mode.

Recommendation

Reword the paragraph as follows:

"3.2.1.1.6 Multi-Speed Trim Actuator. Multi-speed trim actuator systems shall be designed to preclude runaway or inadvertent operation in any trim mode. A trim disconnect switch shall be provided to permit the simultaneous disconnect of all trim modes except for any manual or emergency trim modes that may be used. These requirements shall also apply to single speed pitch trim systems.

Requirement

3.2.1.1.7 FCS Control Panel. The FCS control panel shall provide the pilot with the integrated means to select the MFCS and AFCS functions.

Comparison

On the C-5A aircraft the AFCS and flight augmentation control panels are located on the center console control panel as shown in Figure 4 (3.2.1). The Pilot Assist Cable Servo (PACS) switching is located on the center overhead panel shown in Figure 1 (3.2.1.1.7). The manual flight control system control switches and lights are located on the center overhead panel. The position indicators are located on the center instrument panel. The flight control system controls and indicators are as shown in Figure 2 (3.2.1.1.7). By locating these panels central to the pilots, they have the means to select MFCS and AFCS functions as desired.

Discussion

The requirement is a valid and straightforward one with which compliance can easily be demonstrated. The requirement is valid for future transport aircraft.

Recommendation

Retain the requirement as stated.

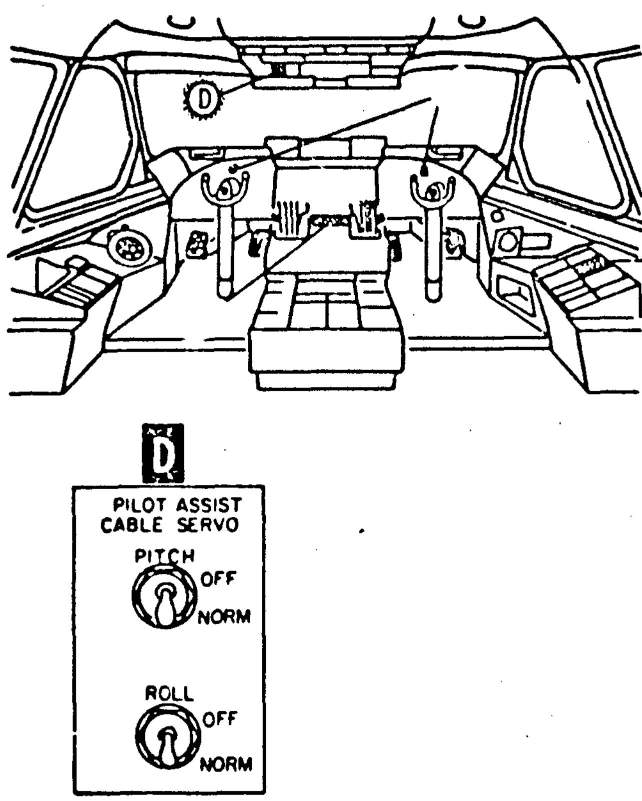


FIGURE NO. 1 (3.2.1.1.7) C-5A PACS CONTROL

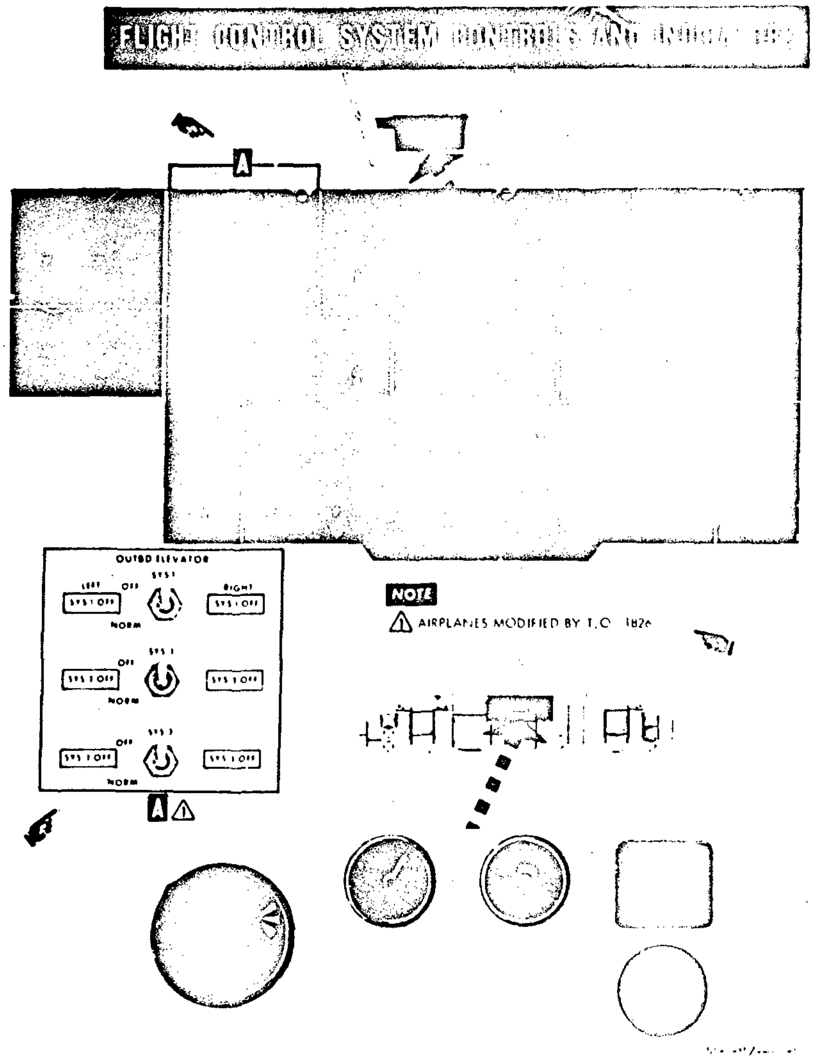


FIGURE 2 (3.2.1.1.7). FLIGHT CONTROL SYSTEM CONTROLS AND INDICATORS

Requirement

3.2.1.1.8 Normal Disengagement Means. Means for disengagement of all modes of the AFCS shall be provided which are compatible with the requirements of 3.1.9.6.

Comparison

The C-5A AFCS disengagement and re-engagement can be achieved in a positive manner under all normal flight conditions. Fail safe in-flight disengagement of the AFCS modes can be manually achieved by pressing the appropriate engage switch a second time, by depressing either control wheels disengage switch or by switching the "MASTER POWER" off. Automatic disengagement occurs when a failure is detected by the monitoring system and is indicated to the pilots through the annunciation system. The AFCS design is such that automatic disengagement or power failure leaves the affected system in the safest mode. Failure in the engage/disengage circuitry causes automatic disengagement and indication annunciation.

Examples of the AFCS control panel and annunciator panel are shown on Figure No. 4 (3.2.1). Additional discussion of the AFCS disengagement modes are noted in the validation of paragraphs 3.1.3.3.2 and 3.1.5.3.2.

Mode compatibility of the AFCS is achieved by interlocks and mode compatibility logic designed to provide safe and efficient FCS operation which is aided by illumination of "mode available" indication lights.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design. The design can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.1.1.9 Preflight Test Controls. Additional controls shall be provided in the cockpit for initiating and controlling the progress of preflight tests, where necessary.

Comparison

The C-5A provides means for preflight testing of both the MFCS and AFCS and associated systems (eg. hydraulic systems) in the cockpit. In the case of the AFCS, through the use of the built-in test equipment (BIT) either a manual or automatic test of the system can be performed. For this aircraft, the majority of system preflight testing is performed by the flight engineers rather than the pilot.

Discussion

The requirement as stated is practical and compliance can be easily demonstrated. The requirement has the proper amount of stringency in that it calls for the controls placement in the cockpit, but does not restrict them to being accessible by the pilots. No changes are necessary for this requirement to remain valid.

Recommendation

Accept as is.

Requirements - Not applicable.

3.2.1.2 Pilot Controls for Rotary-Wing Aircraft

3.2.1.2.1 Interconnection of Collective Pitch Control and Throttle(s) for Helicopters Powered by Reciprocating Engine(s)

3.2.1.2.2 Interconnection of Collective Pitch Control and Engine Power Controls for Helicopters Powered by Turbine Engine(s)

3.2.1.2.3 Alternate or Unconventional Controls

3.2.1.3 Pilot Controls for STOL Aircraft

Requirement

3.2.1.4 Pilot Displays

3.2.1.4.1 FCS Annunciation. The FCS control panel or associated panels shall provide means to display:

a. AFCS engaged
b. Mode engaged
c. That automatic mode switching has occurred, if required
d. Preselected values for selectable mode parameters

Comparison

The C-5A aircraft employs the AFCS control panel, Figure 4 (3.2.1), the FCS panel, Figure 3 (3.2.1.1.7), and the annunciator panel, Figure 1 (3.2.1.4.1) in the cockpit. Through lighted indicators on these panels, the pilots can tell which FCS modes (including AFCS) are engaged and which are available for engagement in the present configuration. They are also given appropriate lighting to indicate if automatic switching has occurred. The preselected values for Mach, airspeed and altitude are displayed on their respective vercical scale instruments. These selectable mode parameters can be varied by the pilots by use of the slew switches located on the AFCS control panel.

Discussion

The C-5A demonstrates full compliance with this requirement. The requirement is in general a good one which can be practically demonstrated. The requirement calls only for the presence of displays leaving the quality and method of implementation to the individual contractor. No changes should be made to the requirement for transport type aircraft.

Recommendation

Accept as is.

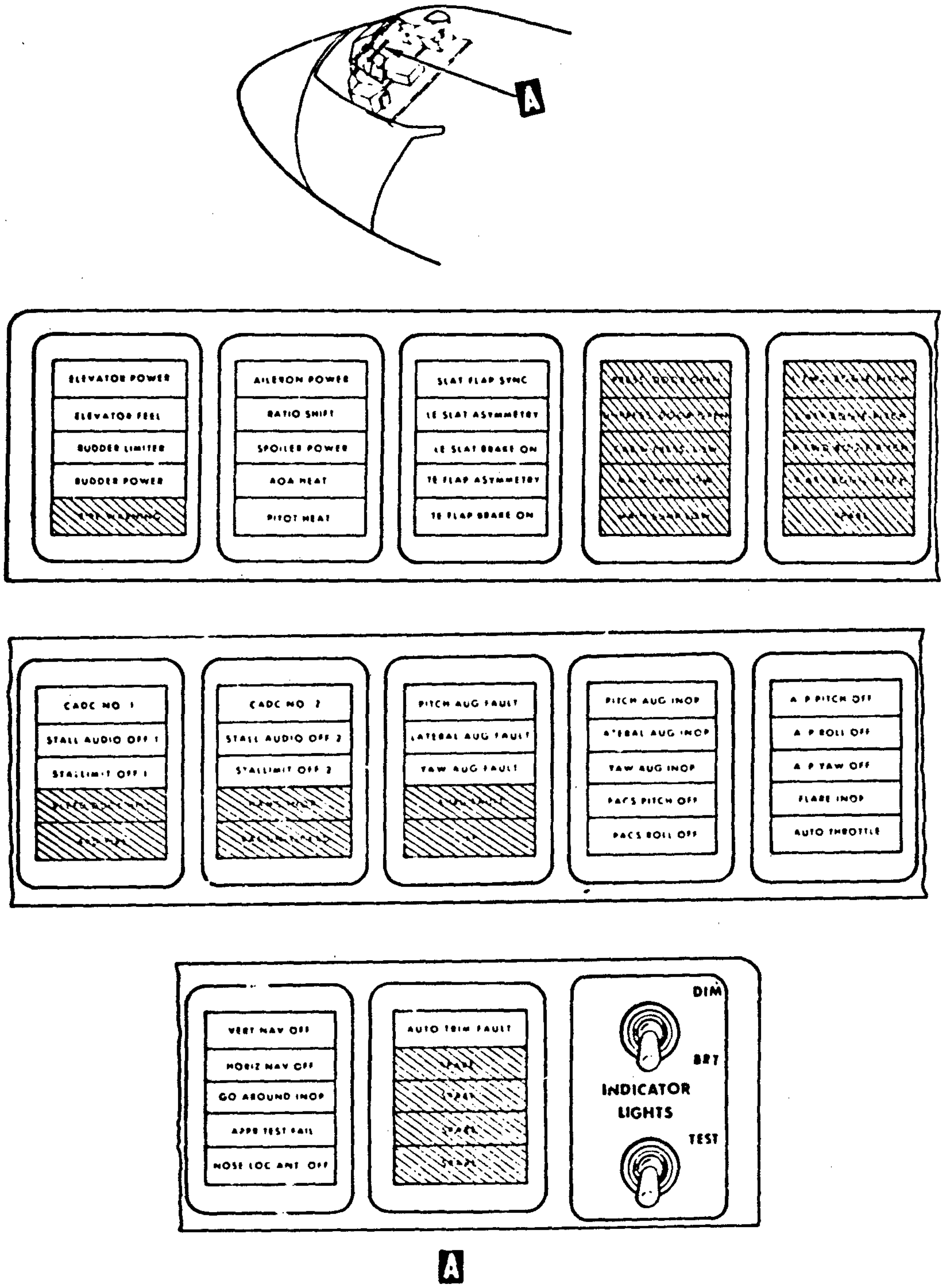


FIGURE NO. 1 (3.2.1.4.1) FLIGHT ANNUNCIATOR PANEL

Requirement

3.2.1.4.2 FCS Warning and Status Annunciation. FCS warning and status annunciation shall be provided in the cockpit. Annunciation shall be designed to clearly indicate the associated degree of urgency.

a. First degree - Immediate action required (warning may be audible).

b. Second degree - Caution, action may be required.

c. Third degree - Informational, no immediate action required.

A panel comprising means for displaying first degree annunciations shall be located within the normal eye scan range of the command pilot. A first degree warning or status indication, which applies only to a particular mode or phase of flight, shall be inhibited or designed to clearly indicate a lesser degree or urgency for all other modes of phases of flight.

Comparison

The C-5A is equipped with a "master caution system" which provides a centrally located method for monitoring all FCS caution and warning indicators in the cockpit. The system consists of amber master CAUTION and master AUTO lights located in front of the pilot and copilot on the main instrument panel and an annunciator panel which contains the indicator lights that can illuminate the caution lights. The annunciator panel lights are white. The pilot's instrument panel is shown in Figure 1 (3.2.1.4.2), and the annunciator panel is shown in Figure 1 (3.2.1.4.1). When any of the annunciator caution lights come on, except those on the last three rows to the right on the panel, the master CAUTION lights on the pilot's and copilot's panels will also light. When any of the annunciator caution lights in the last three rows on the right of the panel come on, the master AUTO lights on the pilot's and copilot's instrument panels come on. This set of annunciator lights warn of failures in the automatic flight control system. The master caution system provides indication of which is the latest malfunction signal if a second malfunction occurs after a first has been indicated. A first malfunction causes the annunciator light to flash. The flashing is changed to a steady light by depressing either master light to reset the system and turn off the master lights. The annunciator light will remain on until the condition has been corrected. If a second malfunction occurs, the applicable annunciator light will flash and the master lights will come on again. However, if a second malfunction occurs in a system where the annunciator light is already indicating a failure, no new warning will be given except for the SAS where the second failure will disengage the system and illuminate the SAS INOP light for that axis on the annunciator panel.

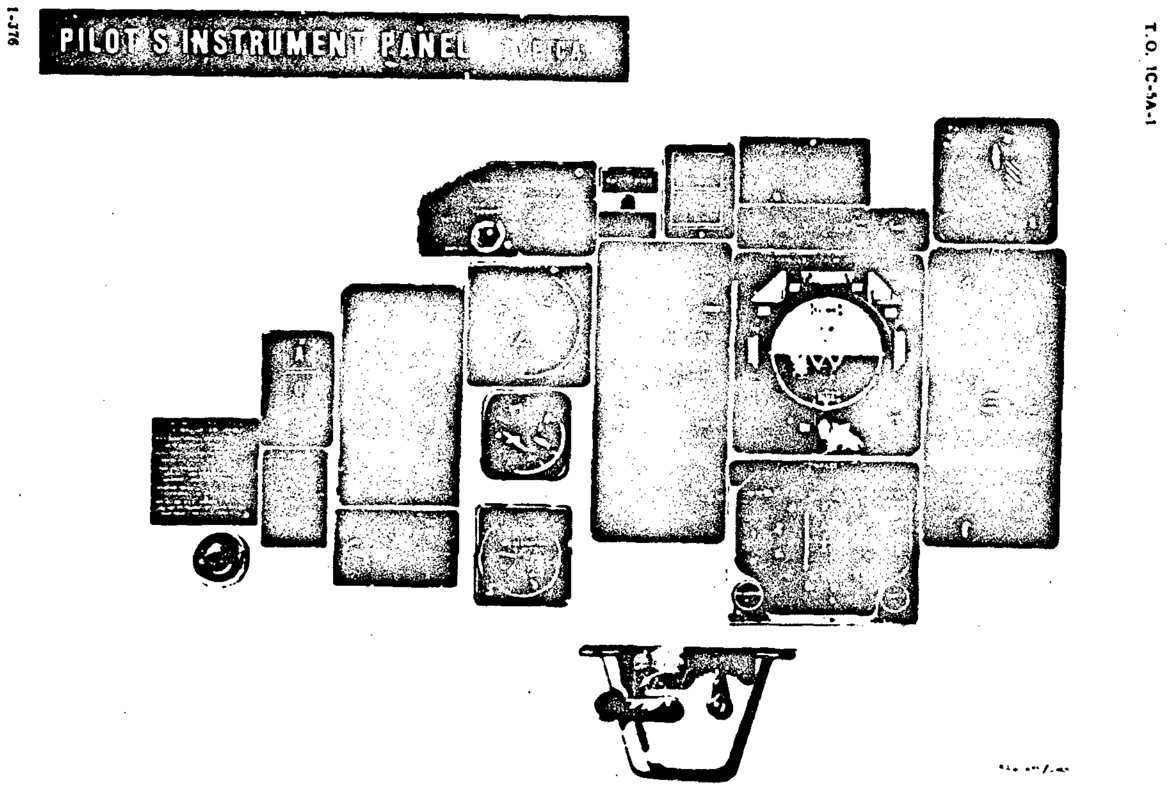


FIGURE 1 (3.2.1.4.2). PILOT'S INSTRUMENT PANEL

The C5A also uses green lights to indicate mode availability or normal operation. White lighting is used for legends for example on the AFCS control panel to indicate which mode is controlled by the switch. Red is used for first degree application such as for the fire warning lights. The C5A meets the intent of this paragraph by using color coding of the warning and static lights to indicate degree of urgency.

Discussion

This requirement is applicable to present and future aircraft and compliance can be demonstrated.

Recommendation

Retain the requirement as stated.

Requirement

3.2.1.4.2.1 Preflight Test (BIT) Status Annunciation. If BIT is used, this display shall:

a. Indicate the progress of the preflight test

b. Instruct the crew to provide required manual inputs

c. Indicate lack of system readiness when failure conditions are detected

Comparison

The BIT procedure on the C-5A has the option of being performed either automatically or manually by an operator. Each step of the test sequence is displayed in binary form by test lights on the face of the computer. If a failure is detected during the test, the sequence is automatically stopped.

This point at which the sequence is stopped is represented in binary form by the lamps that are presently lit. An instruction decal above the test lamps lists applicable documentation which may be consulted to isolate the specific component at fault. The test may then be continued by depressing the BITE switch again. Detail set-up requirements and follow-up instruction for each failure or combination of failures are also provided. The C-5A satisfies each phase of the requirement.

Discussion

The requirement is not restrictive or stringent in making compliance clearly demonstrable and leaving implementation techniques to the designer.

This requirement is applicable to present and future aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.1.4.2.2 Failure Status. Failure warnings shall be displayed to allow the crew to assess the operable status of redundant or monitored flight control systems. Automatic disengagement of an AFCS mode shall be indicated by an appropriate warning display. Manual disengagement by the crew shall not result in warning annunciation.

Comparison

The C-5A displays failure warnings and automatic mode switching of monitored systems on the annunciator panel and AFCS panel shown in Figures 4(3.2.1) and 1(3.2.1.4.1). Manual disengagement does not result in a warning; however, in some modes a display will appear which indicates that a mode is inoperable. The requirement is satisfied with the C-5A's failure status annunciator. The elevator and aileron/spoiler control systems are mechanically redundant flight control systems, but no attempt was made to provide flight station failure warnings.

Discussion

It was implied that this requirement is referring to electronically redundant systems by referring to the "users guide" and previous paragraphs (3.2.1.4.1). The first portion of the requirement, if the above implication is true, can be practically demonstrated. The requirement is not valid for future transport aircraft. The last sentence of the requirement is too restrictive in that it prohibits warning annunciation of accidental or inadvertent disengagement of systems affecting safety of flight. Future aircraft may require SAS operation to assure at least level III flying qualities.

Recommendation

Revise the requirement as follows:

> 3.2.1.4.2.2 Failure Status. Failure warnings shall be displayed to allow the crew to assess the operable status of redundant electronic flight control systems or monitored flight control systems. Automatic disengagement of an AFCS mode shall be indicated by an appropriate warning display. Manual disengagement by the crew of systems not involved in flight safety shall not result in warning annunciation.

Requirement

3.2...4.2.3 Control Authority Annunciation. If available manual control authority can be reduced below the level required for maneuver control by a function such as automatic trim or stability augmentation, pilot displays shall be provided to indicate available control authority for essential and flight phase essential FCS. Warning shall be provided if remaining manual control becomes critical.

Comparison

The C-5A augmentation systems are fail passive type systems. The systems utilize median selectors to reduce the effects of failures on the system operation. Four median selectors and four servo amplifier/feedback loops are utilized in the output channel. Two of the four output channels are used for Channel A and two for Channel B. Under normal conditions, only Channel A is used to drive the surfaces. In the event of a failure which affects the output of Channel A, then Channel B would be automatically switched in and Channel A disabled. When a second failure occurs which affects the output of Channel B, the channel will be disabled, the SAS system will disengage and appropriate warning lights will illuminate. With the system there is no control authority degradation due to SAS operation that will affect proper operation of the aircraft.

Automatic trimming of the C-5A stabilizer is used during autopilot operation. The system is monitored to insure that no probable malfunction will occur that is not announced to the pilot. If an elevator deflection in excess of the trim threshold prevails for three seconds or longer, and the monitor circuit does not detect a change in the stabilizer position transmitter, the AUTO TRIM FAULT annunciator will light, and the automatic trim system will be disabled. Conversely, if a change in the stabilizer position signal persists for three seconds when elevator deflection is below the trim threshold, the AUTO TRIM FAULT light will come on and the auto trim will be disabled. Under unusual conditions, it is possible (though unlikely) that the automatic trim will drive the stabilizer to its limit switches. Should this occur the AUTO TRIM FAULT will illuminate, alerting the pilot to the fact that the aircraft is out of trim. Whenever the AUTO TRIM FAULT comes on and remains on, it must be assumed that the aircraft may be out of trim, and the pitch autopilot should be disengaged while holding the control wheel to restrain any sudden return to neutral.

The C-5A has been designed so that no probable failure will reduce control authority beyond that required for safe flight. Three trim indicators are located in the cockpit to indicate to the pilot the amount of trim being used. There is no provision to compute the amount of control authority remaining during any trim or augmentation operation. The C-5A manual control system authority cannot be reduced to a critical level by normal functions. The C-5A does not meet the requirement to display available control authority.

Discussion

This requirement is too stringent and probably impractical. To obtain control authority for indication to the pilot would require a computer which would monitor airspeed, Mach, altitudes, attitude, center of gravity, all surface deflections, all trim and or other systems operation which affected surface deflection. The computer would then be required to compare the surface deflection capability at any time with that required for maneuver control for that particular phase of flight and determine if sufficient surface deflection was left and then display this to the pilot. To date no accurate method of automatic measuring center of gravity on large transport type aircraft in flight has been used.

Other requirements of MIL-F-9490D cover this area of safe operation and it is felt that this paragraph is unnecessary. Some of the paragraphs are 1.2.2, 1.2.3, 3.1.3.2, 3.1.3.5, 3.1.9, 3.2.1.4.4, etc. If these requirements are met then the basic objectives of this requirement are met in a practical manner.

Recommendation

Delete the requirement

Requirement

3.2.1.4.3 Lift and drag device position indicators. Indicators shall be provided in the cockpit to indicate to the pilot(s) the position of each lift or drag device having a separate control. They shall also indicate the correct takeoff, enroute, approach, and landing positions; and, if any extension of the lift and drag devices beyond the landing position is possible, the indicators shall be marked to identify the range of extension. In addition, an indication of unsymmetrical operation or other malfunction in the lift or drag device systems shall be provided whenever necessary to enable the pilot(s) to prevent or counteract an unsafe flight or ground condition.

Comparison

The C-5A high lift system as shown in Figure II-8 consists of a leading edge slat system and a trailing edge flap system, both controlled by a single control handle. The slat system is two-position in that the slats are stopped only in the extended or retracted positions. Although the flap system can be set at any position between Retracted and Landing (fully extended), the flap handle has two normally used positions. These are detented and are identified as takeoff and takeoff/approach.

The slat position indicator is a rectangular three position flag-type indicator which displays the word "Retracted" if the slats are fully retracted, the word "Extended" if the slats are fully extended and presents a crosshatched bar when the slats are traveling between the extended and retracted positions. The flap position indicator is a round dial-type indicator that is graduated in terms of percent flap extension. The word "Up" is printed at the zero-percent extension position and the word "Down" is printed at the 100-percent extension position. "Down" corresponds to the detented landing flap handle position. The slat and flap position indicators are located to the right of center on the main instrument panel.

The indicator provisions for the slats and flaps on the C-5A have proven adequate and are generally in compliance with the intent of the requirement. However, the flap position indicator does not accentuate takeoff and takeoff/approach positions at the percent extensions corresponding to the equivalent flap handle detented positions. Consequently, the flap position indicator is not in full compliance with the requirement.

A slat and flap asymmetry detection system is used to sense or detect failures in either system that might result in asymmetric operation. The asymmetry sensors are located at the extreme outboard ends of the torque tubes in each wing. If any asymmetry occurs in either the slat or flap system, either a slat or flap asymmetry warning light is illuminated and a corresponding slat or flap "Brake-On" warning light is illuminated in the flight station. The warning lights are located on the annunciator panel at the bottom of the instrument panel. The requirement for indication of slat and flap unsymmetrical operation or other malfunction is therefore satisfied by the C-5A slats and flaps asymmetry detection system.

Discussion

The flap position indicator used on the C-5A is a GFAE part so the additional markings for takeoff and takeoff/approach would make it a non-standard and therefore a more expensive unit. However, the requirement is reasonable and full compliance would produce a better position indicator since the pilots would not have to commit to memory the percent extension which corresponds to the flap handle detented positions. The pilot's workload would therefore be reduced. Consequently, the stringency is justified for future transport aircraft.

Leading edge slat and flap systems normally have two positions, retracted and extended. For such systems, it is unnecessary to use an indicator which displays percent extension. Consequently, revision to this requirement to properly cover two position high lift systems would be appropriate.

Recommendation

1. Add as second sentence to requirement 3.2.1.4.3:

Lift devices such as leading edge slats or flaps normally having two positions, such as extended or retracted, may utilize a three position indicator.

2. Revise existing second sentence of requirement 3.2.1.4.3 to read:

Lift and drag devices normally having more than two positions shall also indicate the correct takeoff, enroute, approach, and landing positions; and, if any extension of the lift and drag devices beyond the normal maximum landing position is possible, the indicators shall be marked to identify the range of extension. In addition, an indication of unsymmetrical operation or other malfunction in the lift or drag device systems shall be provided whenever necessary to enable the pilot(s) to prevent or counteract an unsafe flight or ground condition.

Requirement

3.2.1.4.4 Trim Indicators. Suitable indicators shall be provided to:

a. Indicate the position and the range of travel of each trim device.

b. Indicate the direction of the control movement relative to the airplane motion.

c. Indicate the position of the trim device with respect to the range of adjustment. (Trim devices such as the magnetic brake used in helicopters to instantaneously relieve pilot's control forces by changing the feel force reference to zero at the control position held by the pilot at the time the trim switch is activated shall not require separate trim indicator.)

d. Provide pilot warning of trim failures which could result in exceeding the State III requirements of 3.1.3.3.4.

Aircraft which require takeoff longitudinal trim setting in accordance with cg location shall have suitably calibrated trim position indicators. Where suitable, trim indicators shall be in accordance with MIL-I-7064. In aircraft requiring quick takeoff capability or certain single pilot aircraft, which use a single trim setting for all takeoff conditions, a trim for takeoff light shall be provided.

Comparison

C-5A pitch, roll and yaw trim position indicators, as shown on Figure No. 1 (3.1.3.5) , are used to display to the pilots calibrated position, range of travel, direction of the control movement relative to aircraft motion and the trim position with respect to the range of adjustment. The aileron trim indicator is a dual pointer type indicator with a trim range of plus and minus 10 degrees. Aileron trim may be accomplished with the aileron trim knob or with a switch, both of which are located on the center console. Operation of the aileron trim knob provides a simultaneous electrical signal to the left hand and right hand aileron trim actuators. Operation of the trim switch provides an electrical signal to only one trim actuator at a time. Simultaneous operation of both aileron trim actuators cause the left and right aileron trim indicator pointers to move as a unit in a clockwise or counter-clockwise direction depending upon the trim direction commanded. If the switch is used to affect trim, only the associated left aileron or right aileron pointer will move to display the amount of trim.

The rudder trim indicator is a dial type indicator having a single pointer and displaying a trim range of plus and minus twelve (12) degrees. Rudder trim is accomplished by simultaneously depressing two switches which provide power and ground signals to the trim actuator. The switches are three position (nose left, off, nose right) toggle switches spring loaded to the off position. The upper and lower rudder surfaces are trimmed simultaneously as if the input were due to pedal deflection. The trim actuator provides plus and minus eleven (11) degrees trim authority at a rate of one (1) degree per second and trim position is displayed on the rudder trim position indicator located on the center instrument panel.

In addition to the rudder trim provisions described above, an emergency rudder trim control provides the pilot with plus and minus twenty (20) degrees of upper and lower rudder trim authority. This emergency trim control is provided through the use of the Yaw Augmentation Manual Trim control knob located on the Flight Augmentation panel. Emergency rudder trim control indication is provided by a scale marked on the panel beneath and around the periphery of the control knob. A guarded switch to the right of the control knob must be moved from the "Off" position to the "On" position before the emergency mode becomes operational.

The pitch trim indicator is a dial type indicator having a single pointer and displaying a trim range of fourteen (14) degrees aircraft nose up and six (6) degrees aircraft nose down. In addition, there is a motion or rate indicator within the face of the indicator to provide a more positive indication of horizontal stabilizer movement. The indicator is located on the main instrument panel and gives the pilots a visual indication of the stabilizer position in degrees up or down from its neutral position. Pitch trim control is normally accomplished by means of pitch trim switches on the outboard grip of the control wheels. Trim may also be accomplished by means of the Alternate Pitch Trim Switches located on each side of the aft center console.

Except for the rate indicator on the face of the pitch trim indicator, the trim indicators described above do not incorporate separate provisions for warning the pilots of trim system failures resulting in inadvertent trim. It is not likely that the slow movement of the indicator pointers would direct pilot attention to trim runaway. However, trim runaway would be immediately discernible through the pilot's controls due to the upset in aircraft trim. In conclusion, the trim indicators comply with the intent of the requirement.

Discussion

The requirement as written in reasonable and can be practically demonstrated. The pitch trim indicator used on the C-5A meets the requirement for pilot warning of trim failures which could result in exceeding the State III requirements of 3.1.3.3.4 by incorporating a rate indicator as previously described. It was considered necessary because the rate the pointer moves corresponding to the normal trim rate is not fast enough to gain the pilot's attention in the event of inadvertent trim.

Recommendation

The requirement is accepted as is.

Requirement

3.2.1.4.5 Control Surface Position Indication. Indicators shall be provided in the cockpit for all control surfaces whose positions are indicative of potential flying qualities below Level 3, when the cockpit controls do not provide a positive indication of long term or steady state control surface position, or where the effect of control surface positioning is not readily detectable by other means.

Comparison

The position of the primary control surfaces on the C-5A is not displayed in the cockpit by the use of position indicators. The C-5A primary control system is designed to accept the loss of any two hydraulic systems and still permit control about any axis. The decision was made during the project design effort on the C-5A that this hydraulic redundancy together with the mechanical and control surface redundancy, as shown on Figure No. 1 (3.1.8.1) did not indicate the need for surface position indicators. However, it is clear that the C-5A does not meet the requirement of this paragraph.

Discussion

It is recognized that the safety of an all-weather landing system would be significantly enhanced by the use of position indicators. In addition, failures and malfunctions can more readily be isolated and managed by the crew--particularly where multiple control surfaces are used for control about each axis. Hence, it is considered to be a good requirement for aircraft control systems involving the use of multiple control surfaces and/or series trim.

Recommendation

Retain the requirement as stated.

Requirement

3.2.2 Sensors. Sensors shall be installed in locations which allow adequate sensing of the desired aircraft and flight control system parameters and which minimize exposure to conditions which could produce failures or undesired output signals.

Comparison

The C-5A FCS sensors have been designed and located in a manner which provide signals for proper operation of all the related systems under all the applicable operating conditions of the mission requirements and under the influence of the specified induced and natural environments as defined under specification paragraph 3.1.9. The sensors are located in a manner which affords protection from environmental conditions and is accessible for removal, inspection and easy maintenance.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design. The requirement can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.3 Signal Transmission

3.2.3.1 General Requirements

3.2.3.1.1 Control Element Routing. Within the restrictions and requirements contained elsewhere in this specification, all portions of signal transmission subsystems, including cables, push-pull rods, torque tubes, and electrical wiring shall be routed through the airplane in the most direct manner over the shortest practical distances between points being connected. Protection from use as steps or handholds shall be provided.

3.2.3.1.2 System Separation, Protection and Clearance. Where redundant cable, pushrod, or electrical wiring are provided, they shall be separated as required to meet the invulnerability requirements of 3.1.9. Advantage shall be taken of the shielding afforded by heavy structural members, existing armor plate, or other equipment for the protection of important components of the control systems. Clearance between flight control system components and structure or other components shall be provided as necessary to insure that no probable combination of temperature effects, air loads, structural deflections, vibrations, build-up of manufacturing tolerances, or wear, can cause binding or jamming of any portion of the control system. In locally congested areas only, the following minimum clearances may be used after all adverse effects are accounted for:

a. 1/8-inch between static elements except those within an LRU where closer clearances can be maintained or where contact cannot be detrimental.

b. 1/8-inch between elements which move in relation to each other and which are connected to or are guided by the same structural or equipment element(s) except those within an LRU where closer clearances can be maintained or where contact cannot be detrimental.

c. 1/4-inch between elements which move in relation to each other and which are connected to or are guided by different structural or equipment elements.

d. 1/2-inch between elements and aircraft structure and equipment to which the elements are not attached.

3.2.3.1.3 Fouling Prevention. All elements of the flight control system shall be designed and suitably protected to resist jamming by foreign objects.

Comparison

The C-5A control systems were designed to meet the redundancy, separation, protection, and clearance requirements of the Contract End Item (CEI)

Specification CP40002-6B. Where redundant system elements were required they were separated and protected to meet the applicable invulnerability requirements. The clearances between the control system moving elements and the adjacent structure or equipment were implemented with the design goal of "no functional degradation, considering such parameters as structural deflections, vibration, temperature, tolerances, etc."

The intent of this design goal was satisfied in meeting the C-5A design Contract End Item requirements for FCS redundancy, operation, protection, and clearance.

The C-5A control system mechanical signal transmission is routed from the pilots controllers through a series of cable, bellcrank, push-rod arrangements to the control valves on the fully powered surface actuators. Figures 1, 2, and 3 (3.2.3.1.1) describe the control element routings of most of the C-5A control systems. The electrical/electronic control elements for the control actuators or related components receive their signals through electrical wires.

In both mechanical and electrical systems the signal transmitting elements have been routed in the most direct manner in the interests of both weight savings as well as efficiency.

Considerable emphasis was placed on signal accuracy of the transmitting elements. The mechanical system used "Lock Clad" cable, close tolerance bolts, and rigid components to minimize system deflection and free play. The routing of the mechanical elements was integrated, where possible, with existing areas adjacent to aircraft structure which afforded the maximum protection. Where dual systems were required, the transmitting elements were separated to the maximum extent possible.

In all of the moving control elements, the minimum clearance requirements were usually exceeded in order to preclude contact with adjacent structures and other equipment. The operating extremes and structural deflections were considered in setting these clearances . Access was provided for maintenance and rigging of all components. All the control elements were designed to provide the redundancy and margins of safety necessary to meet the failure criteria established by the C-5A operational state criteria.

The basic design goal specified that all the control system elements shall be suitably guided, protected by location, <u>or</u> covered to prevent their being fouled accidentally or during maintenance operations. The C-5A design considered the requirement to resist jamming by foreign objects within the design guidelines of weight, budget, maintainability, and reliability. However, the design philosophy for the failure criteria assumed a jam in the system of control elements. Therefore, to meet the operational failure state criteria of the CEI specifications, all the critical control systems

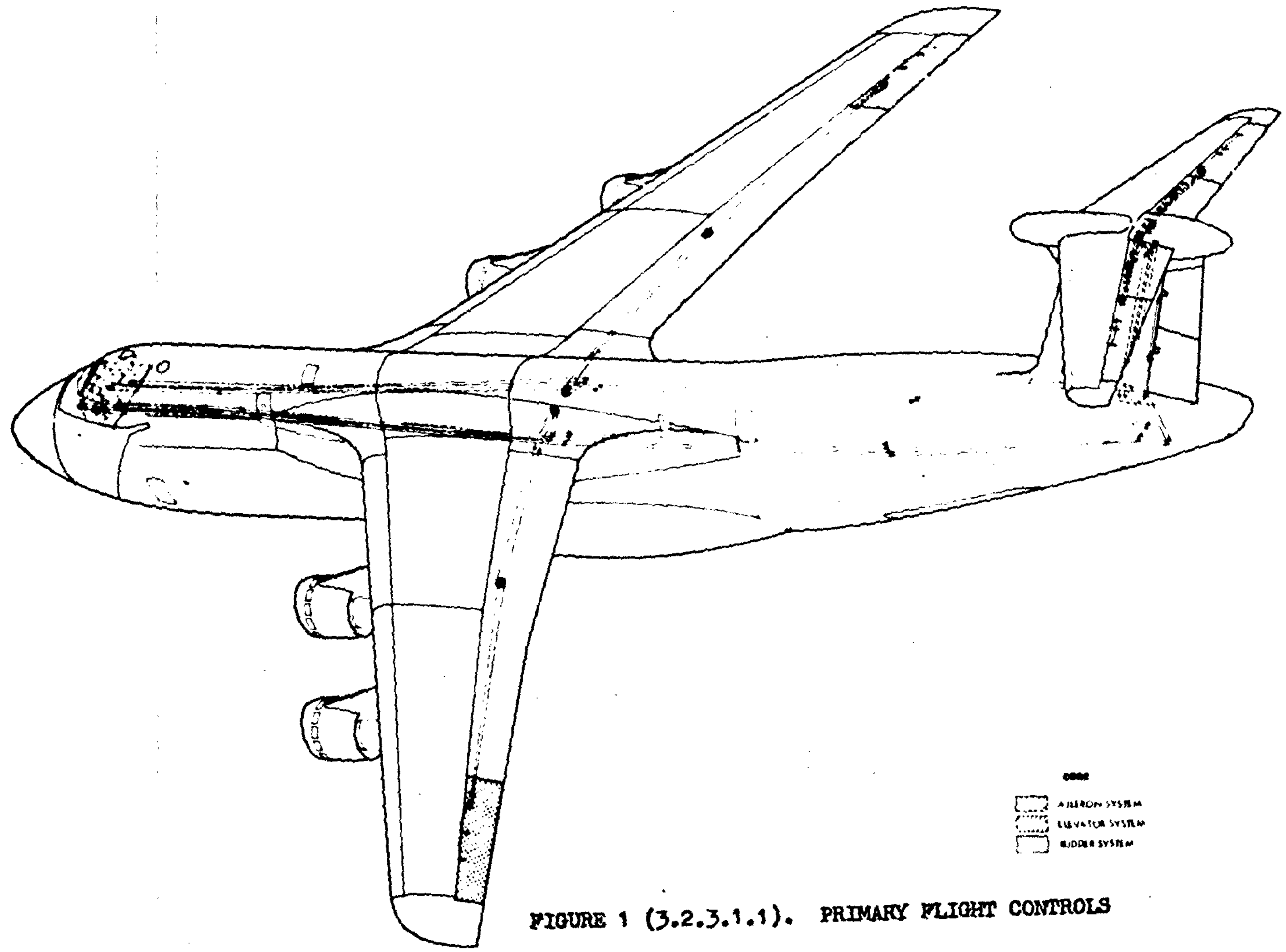


FIGURE 1 (3.2.3.1.1). PRIMARY FLIGHT CONTROLS

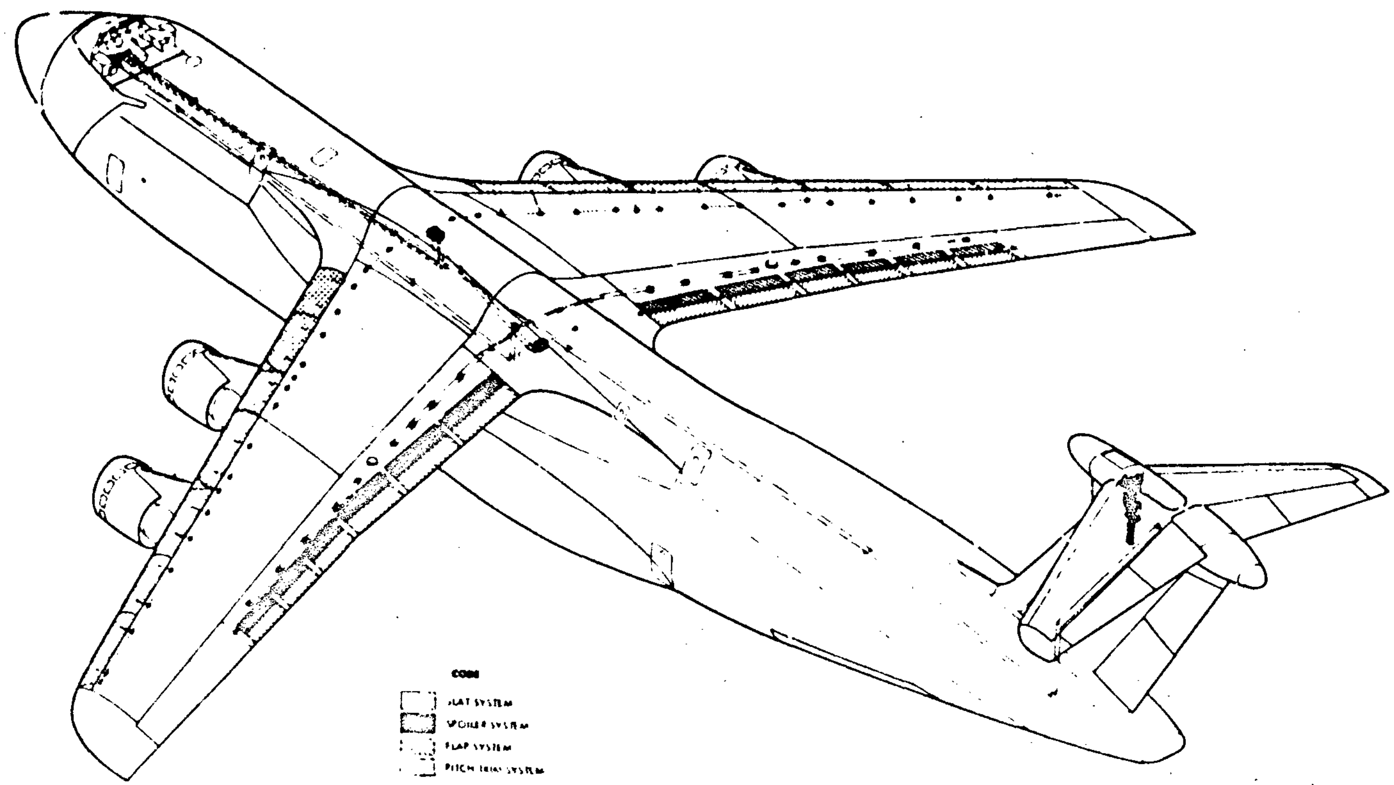


FIGURE 2 (3.2.3.1.1). SECONDARY FLIGHT CONTROLS

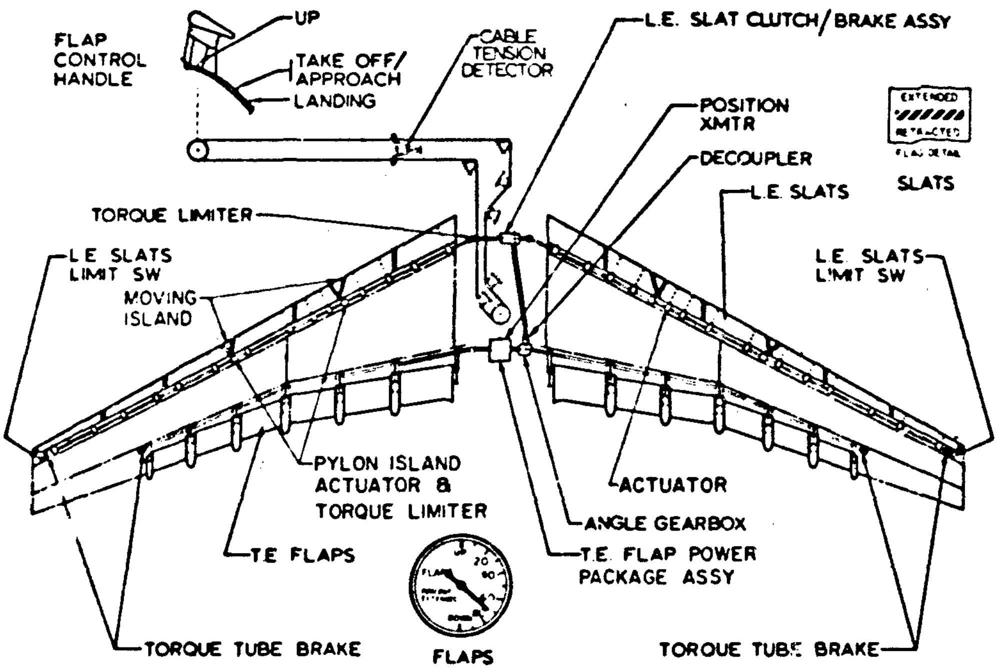


FIGURE NO. 3 (3.2.3.1.1) FLAP SYSTEM - MAJOR ELEMENTS

were provided with redundant control paths. In the event of a jammed system the redundant system was disconnect or "sheared out" from the jammed portion of the system to maintain control of the aircraft.

Discussion

This is a good requirement when it is considered as a design goal. The interpretation in this, as in any general specification, has to be supplemented by the configuration application as well as the design guideline "checklist" as defined in the "Users' Guide." Meeting this type requirement cannot be demonstrated except by abstract, non-quantitative terms by opinion based on experience and/or analysis.

It is suggested that it would be beneficial to develop more specific requirement guidelines from the results of hardware "tradeoff" studies to determine the characteristics of this design goal versus vulnerability, operation, cost, weight, etc., for future inclusion in the "Users' Guide." This design goal is valid and has been satisfied by the C-5A design and should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Additional Data (For "Users' Guide")

Care should be taken either to avoid routing concentrations of mechanical, electrical, or hydraulic control elements and control power sources and to separate and/or shield the systems as required to minimize single failure effects where system concentration might provide significant multiple failure potential. Typically conjested areas in transport aircraft include rear beam of the wings and vertical and horizontal stabilizers.

Requirement

3.2.3.1.4 Rigging provisions. The number of rigging positions shall be kept to a practical minimum. They shall be readily accessible and located where space for the rigging function is available. Installed rigging pins shall be highly visible from the ground or include streamers as specified in 3.1.10.4. Control surface actuator outputs shall not be rig pinned.

Comparison

The intent of this design goal was satisfied in meeting the C5A design Contract End Item requirements. The design emphasized the use of a minimum of rig pin locations, which were generally oriented to major sections of the aircraft. For example, the empennage control systems could be rigged and cleared independently from the rest of the control system. The servicing and maintenance time requirements of the CEI specification assured ready access of the rigging provisions. In applications where the system functions were not completely cycled during maintenance or rigging functions, streamers were used on the rig pins. However, the nature of most control system rig pin locations necessitated rigid system functional checkouts as a final means of checking for rig pin removal.

No C-5A control surface actuator output is rig-pinned.

Discussion

This requirement to minimize the number of rigging positions is good. The C-5A meets the remaining parts of this requirement including those for location and accessability of rigging functions and visibility of rig pins.

Recommendation

Accept the Specification "as is".

Additional Data (For Users Guide)

The initial systems checkout and periodic maintenance of large transport aircraft may be improved by breaking down the rigging into small subsections which are related to the major aircraft structural subsections containing portions of the FCS. The portions of FCS in the wings, horizontal stabilizer, vertical fin and fuselage subsections can be rigged separately before being connected to their adjacent subsections.

The C-5A FCC in the T-tail, horizontal and vertical stabilizer assembly, were completely rigged and functionally tested before the empennage was attached to the fuselage.

Requirement

3.2.3.2 Mechanical Signal Transmission

3.2.3.2.1 Load Capability. Elements of mechanical signal transmission systems subjected to loads generated by the pilot(s) shall be capable of withstanding the loads due to pilot's input limits specified in MIL-A-8865, Section 3.7, Flight Control System Loads, taken as limit loads, unless higher loads can be imposed such as by a powered actuation system or loads resulting from aerodynamic forces. Where higher loads are thusly imposed, they shall be met with the same margins and circumstances as specified in MIL-A-8865.

Comparison

The C-5A flight control systems (FCS) and components were designed to meet the strength requirements of Contract End Item (CEI) specification CP 40002-2, Paragraph 3.4.10.5. The CEI design criteria is essentially the same as the requirements specified in MIL-A-8865, Section 3.7. The dual C-5A FCS components and mechanical system transmitting elements were designed to withstand the maximum limit load resulting from a pilot input load equivalent to 75 percent of the dual pilot effort specified in MIL-A-8865, Section 3.7 when applied at the pilots controller and reacted throughout the control system. The design limit loads exceeded the maximum functional operating loads of the system. All the secondary and other control system elements were designed to the limit load criteria of the CEI specification.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.3.2.2 Strength to Clear or Override Jammed Hydraulic Valves. All mechanical elements which transmit input commands to metering valves of hydraulic servoactuators shall have strength to withstand higher loads, above those for normal valve stroking, required to clear foreign material that may occur in projected usage.

Comparison

The C-5A flight control system (FCS) was designed to the limit load design criteria of the CEI specification for determining the strength of the mechanical signal transmitting elements. In the primary flight control input system that was normally three-fourths of two pilots' effort, which was always much higher than the normal pilot effort to move the full power system control valves and feel system. In addition, all of the hydraulic servo control valves and their related input linkages were designed to a 600 pound limit axial load valve chip shearing force requirement. Override bungees limit the loads which can be applied to less than the 600 pound axial load.

However, it should be noted that the system redundance and jam protection requirements for the C-5A primary flight control system necessitated the use of override bungee springs in most of the input systems. A jam in the C-5A can be overridden or relieved because of strategically located override bungees and shear fuses within the FSC. Also a jammed servo system would be de-energized and control would be maintained through the redundant system.

However, the systems are designed to accommodate the valve chip shearing force requirement where override bungees are not present or where possibly a second failure condition may require this force clearing operation. The actual jam clearing forces available are dependent on the servo valve design and system approach.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.3.2.3 Power Control Override Provisions. Provisions shall be made to permit the pilot(s) to clear or override metering valve jams unless there is sufficient aerodynamic control power from the remaining operative surfaces to override control moments generated by the jammed surface in its most adverse position.

Comparison

The Primary Flight Control System (PFC) for the C-5A has incorporated override bungees at the servo control valve input arm for the Aileron, Flight Spoiler, Inboard Elevator, and Rudder Power Units as shown in Figures II-4, II-5, and II-7.

This application of override springs in the redundant system will permit the pilot to maintain control of the aircraft in the event of a jammed control valve. The bungees have full pilot input stroke capability which permits immediate pilot action to minimize the effects of a hardover servo valve. The pilot may then de-energize the servo containing the jammed valve and, by deflecting the override bungee, maintain control of the aircraft with the remaining servo.

Figure 1 (3.2.3.2.3) shows a typical servo override bungee installation; in this case it is for the Aileron servo.

The input system is also designed to meet the control valve chip shearing strength requirement of 3.2.3.2.2 in the event of a multiple failure condition (i.e., jammed valve and jammed bungee spring).

A degree of protection for a jammed control valve on the primary servo manifold is the dual concentric control valve spool on units with stability augmentation. One spool provides manual control and the other spool controls stability augmentation input. Thus, if either spool becomes jammed either of the remaining spools could afford some degree of control.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

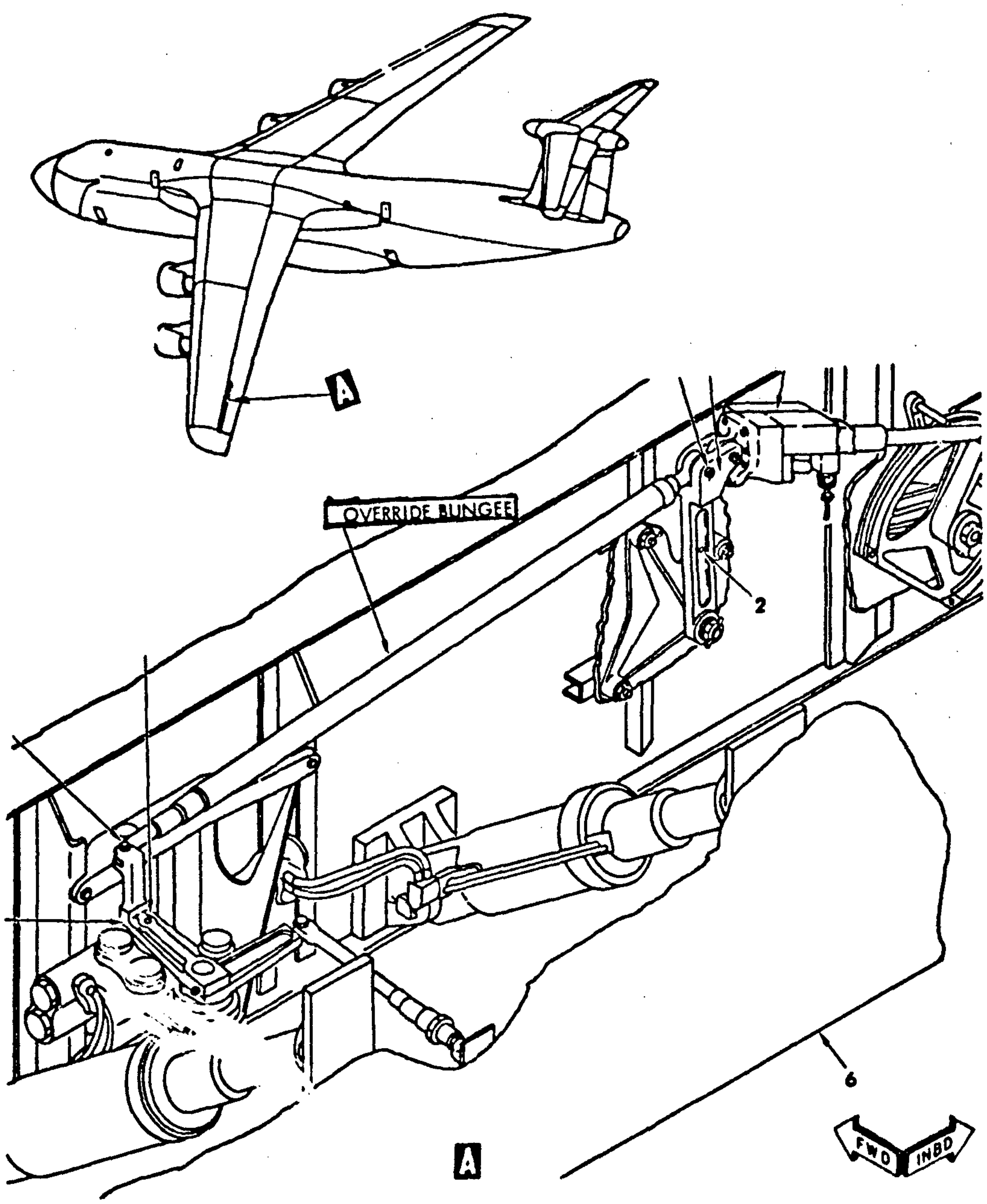


FIGURE 1 (3.2.3.2.3) AILERON OVERRIDE BUNGEE INSTALLATION

Requirement

3.2.3.2.4 Control Cable Installations. Control cable installations shall be designed to accommodate easy servicing and rigging, and the number of adjustments required shall be kept to the practical minimum.

Comparison

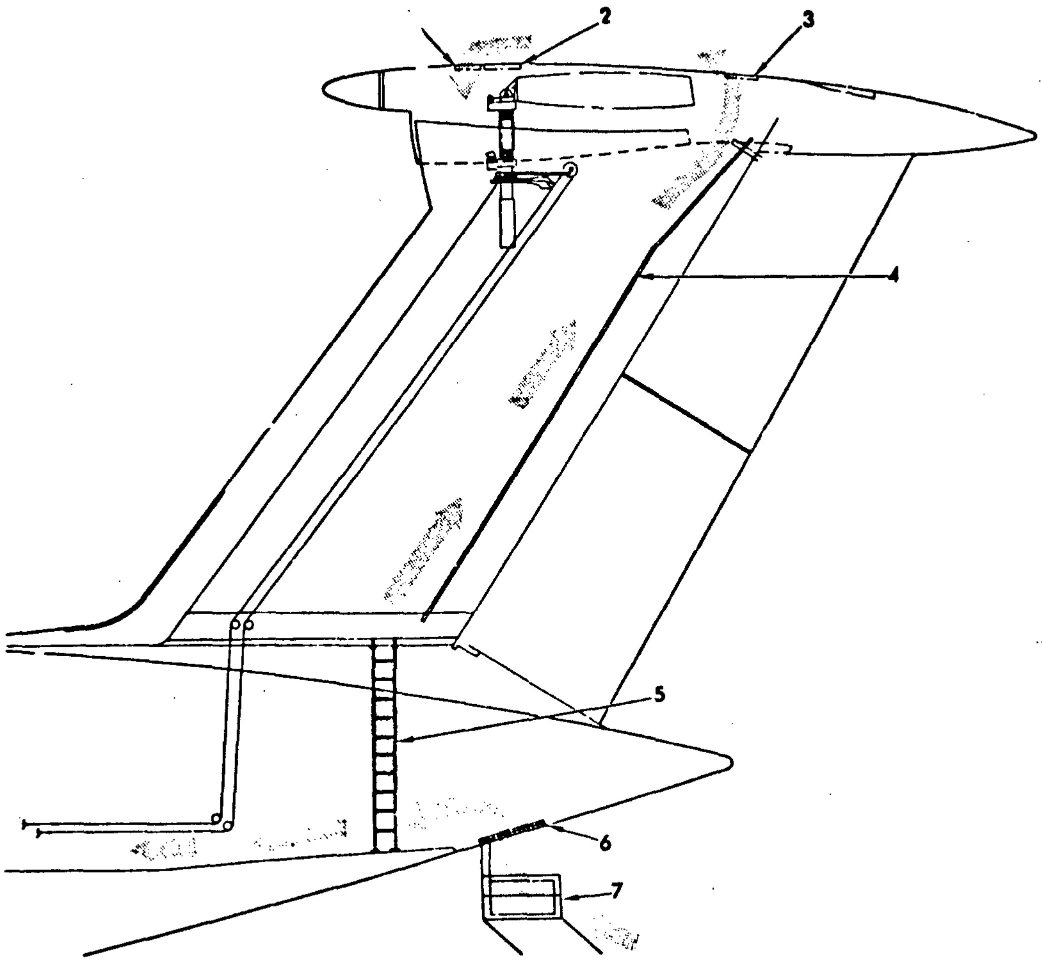
The intent of this design goal was satisfied in meeting the C-5A desig Contract End Item (CEI) requirements. The C-5A flight control system (FCS) able installation was designed to minimize the number of adjustmen s and rig pin locations. The FCS design requirement for easy servicing and minimum maintenance time is assured by virtue of the strict CEI requirements. These maintainability requirements had time values which were met by demonstration to the customer. Figures 1 and 2 (3.2.3.2.4) are typical examples of the C-5A's maintainability access. Figure 1 (3.2.3.1.1) shows the C-5A FCS mechanical control cable runs.

Discussion

This requirement to accommodate easy servicing requires application of good engineering judgement based on experience. This requirement is valid and has been satisfied by the C-5A design and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.



1. PERSONNEL ACCESS DOOR
2. PITCH TRIM ACTUATOR DOOR
3. PERSONNEL ACCESS DOOR
4. SERVICE LADDER
5. SERVICE LADDER
6. PRESSURE VENT AND ACCESS DOOR
7. B-2 STAND

FIGURE 1 (3.2.3.2.4) C-5A AFT BODY AND EMPENNAGE CONTROL ACCESS

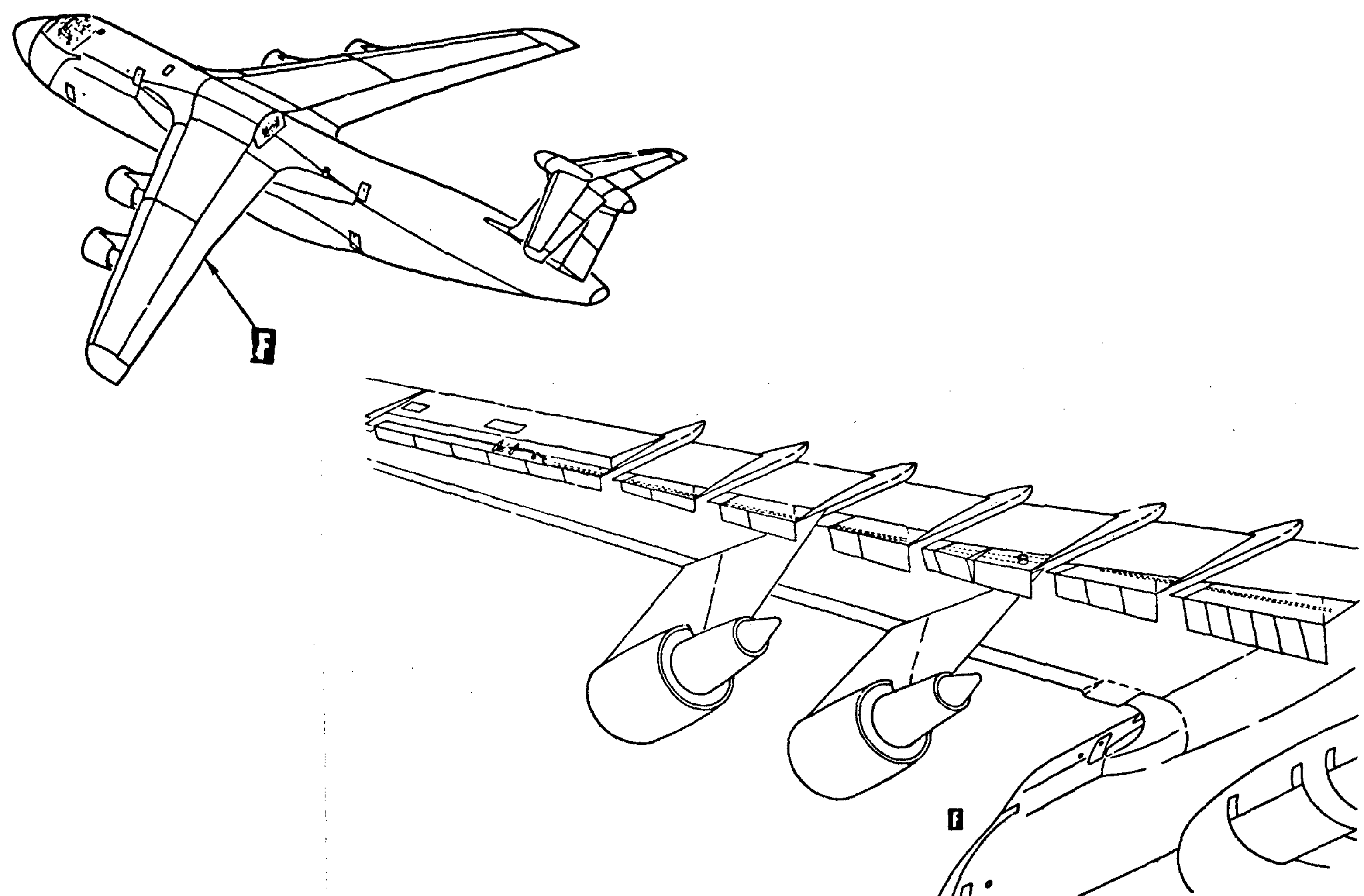


FIGURE 2 (3.2.3.2.4). C-5A AILERON CONTROL ACCESS

Requirement

3.2.3.2.4.1 Control Cable. Cable used for the actuation of flight controls shall be the most suitable of the following types for each application. Use of carbon steel or other type cable not listed below requires procuring activity approval.

a. Flexible nylon-coated corrosion-resisting steel wire rope in accordance with MIL-W-83420.

b. Preformed flexible corrosion-resisting steel wire rope in accordance with MIL-W-83420.

c. Preformed flexible corrosion-resisting nonmagnetic steel cable in accordance with MIL-C-18375.

Comparison

The C-5A mechanical flight control system FCS uses control cables which are preformed flexible corrosion-resisting steel wire rope in accordance with MIL-W-83420. The stainless steel cable was selected because of its superior (to carbon steel cable) properties of fatigue life, environmental protection (thermal and corrosive) and it was a requirement.

In cable applications which were exposed to fatigue fretting, such as where cables have excessive wear due to contact with fairlead rollers, the cable was jacketed with extruded plastic materials. Experience with the C-5A and C-141 has shown a substantial increase in fatigue life with this application. In order to reduce cable stretch, aluminum clad was added to the cable in areas of straight cable runs.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.3.2.4.2 Cable Size. Cable shall be sized to meet the load requirements of the system with ample safety factors to compensate for wear and deterioration where pulleys, fairleads, etc., are encountered. Cable size shall also be adequate in regard to permissible cable stretch, pulley friction values, and other variables which affect system performance. Where substantial loads are carried, cables shall be sized so that limit loads do not exceed 67 percent of the rated breaking strength of the cable and do not exceed the maximum cable limit loads allowed for their pulleys.

Comparison

The C-5A flight control system FCS cables were sized to the Contract End Item requirements and to the limits specified in MIL-Handbook 5.

The primary mechanical FCS was designed to a limit load input minimum of three-fourths of two pilots, applied at the pilots controllers. The cable system limit load strength capability exceeded this requirement. The actual operational load was much less than the design limit load since only the valve, friction, and feel system forces are reacted by the pilot. Therefore, ample safety factors are provided to compensate for wear and deterioration where the cable quadrants, pulleys, fairleads, etc., are encountered. All the pulleys, brackets, quadrants, and cable system components were designed to the maximum load which could be imposed by the application of the maximum specification design limit load at the pilots controllers. The design limit loads of all the FCS cable systems do not exceed 67 percent of the rated breaking strength of the cable. The cable system friction was kept to a minimum by using the large diameter pulleys and minimum pulley wrap angle allowed by the optimum cable run determined by a cost, weight, and performance trade-off study. Cable stretch was minimized by the use of aluminum "Lock Clad" to add stiffness with a minimum of weight.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.3.2.4.3 Cable Attachments. The minimum practical number interconnections shall be used which allow all cable segments to be '. nected manually. Cable disconnects shall be located and designed so that it is physically impossible to misconnect in any manner, either cables in the same system or the cables of different systems. Cable disconnects and turnbuckles shall be so located that they will not hang up or interfere with adjacent structure or equipment or on each other and will not snag on cables, wires, or tubing. Corrosion-resistant steel MS swage-type cable fittings in accordance with MIL-T-781, swaged to form cables assemblies in accordance with MIL-T-6117, shall be used wherever possible. Thimble ends in accordance with MIL-T-5677, attached to cable by splicing and wrapping in accordance with MIL-S-5676, may be used in applications where additional joints are needed to prevent bending fatigue failures. Turnbuckles used in flight control cables systems shall be in accordance with MIL-T-8878. Turnbuckle and fittings shall be designed so that they are not subject to bending forces which can cause fatigue failures. Turnbuckle terminals shall not have more than three thr ds exposed at either end. All turnbuckle assemblies shall be properly safet. d in accordance with MS33736.

Comparison

The number of mechanical interconnections used on the C-5A mechanical flight control cable systems have been minimized in order to provide easier installation and maintenance of these systems. The cable disconnects have been designed and located so that it is physically impossible to misconnect cables in the same or different systems. Figure 1(3.2.3.2.4.3) represents a typical C-5A mechanical flight control cable system. On Figure 1(3.2.3.2.4.3) note that the turnbuckle connections on adjacent cable runs are staggered such that cross connection of the cables is impossible. All cable turnbuckles and connections are spaced and located to provide clearances necessary to assure that no interference or hang up will occur with adjacent structure or equipment.

The cable terminals were MS swage-type, corrosion-resistant steel which were swaged to form cables in accordance with Lockheed's process specifications which satisfy the intent of MIL-T-5677.

MS21251 turnbuckle bodies, per MIL-T-8878, were used in the cable assembly installations. The turnbuckle and fitting installation were designed to preclude the possibility of fatigue damage resulting from any bending forces. The design lengths of the cable assemblies are such that after adjustment of the turnbuckles to acquire the minimum rig load and considering the maximum tolerance effects, all of the threads of the turnbuckle terminals remain within the turnbuckle as shown in Figure 2(3.2.3.2.4.3) A.

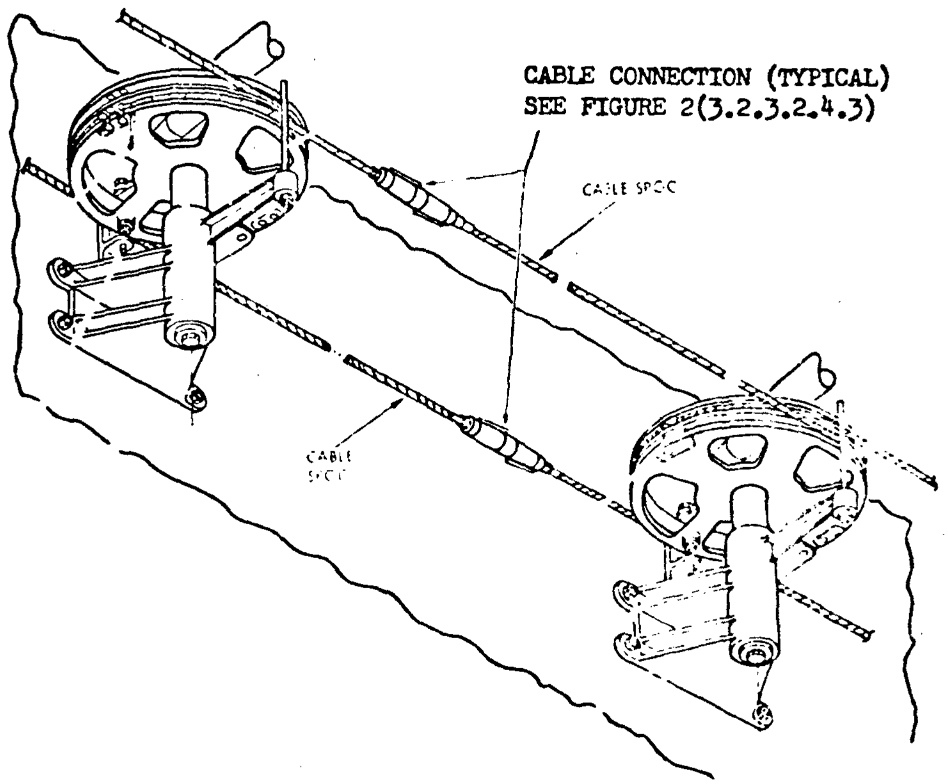
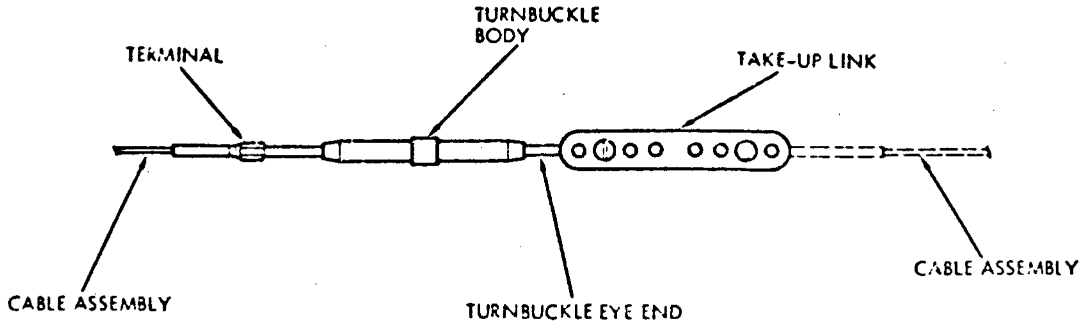
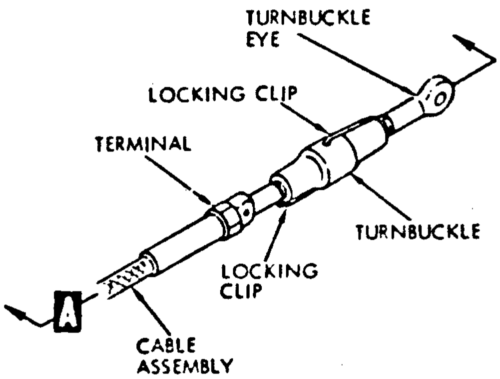
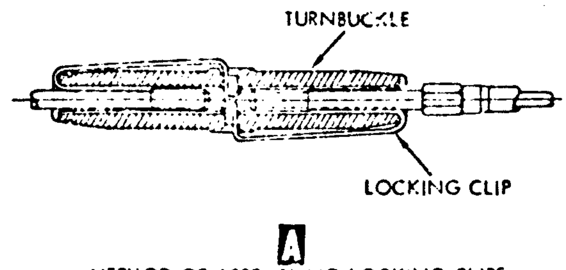


FIGURE NO. 1(3.2.3.2.4.3) C-5A TYPICAL CONTROL CABLE ATTACHMENT





TURNBUCKLE SAFETYING



METHOD OF ASSEMBLING LOCKING CLIPS, TURNBUCKLE BODY AND TERMINALS

NOTE:

1. PRIOR TO SAFETYING, BOTH THREADED TERMINALS SHALL BE SCREWED APPROXIMATELY EQUAL DISTANCE INTO THE TURNBUCKLE BODY. NOT MORE THAN THREE THREADS OF ANY TERMINAL SHALL BE EXPOSED OUTSIDE THE BODY.

2. AFTER THE TURNBUCKLE HAS BEEN ADJUSTED TO ITS LOCKING POSITION, WITH THE SLOT INDICATOR GROOVE ON TERMINALS AND SLOT INDICATOR NOTCH ON BODY ALIGNED, INSERT THE END OF THE LOCKING CLIP INTO THE TERMINAL AND BODY UNTIL THE U CURVED END OF THE LOCKING CLIP IS OVER THE HOLE IN THE CENTER OF THE BODY. PRESS THE LOCKING CLIP INTO THE HOLE TO ITS FULL EXTENT. THE CURVED END OF THE LOCKING CLIP WILL EXPAND AND LATCH IN THE BODY SLOT. TO CHECK PROPER SEATING OF LOCKING CLIP, ATTEMPT TO REMOVE PRESSED U END FROM BODY HOLE WITH FINGERS ONLY, (DO NOT USE TOOLS AS LOCKING CLIP COULD BECOME PERMANENTLY DISTORTED).

3. LOCKING CLIPS ARE FOR ONE TIME USE ONLY, AND SHALL NOT BE REUSED.

FIGURE NO. 2(3.2.3.2.4.3) C-5A CONTROL CABLE ADJUSTMENT AND SAFETYING

All cable assembly turnbuckles are safetied by the use of MS21256 clips as shown in Figure 1 (3.2.3.2.4.3). Figure 2 (3.2.3.2.4.3) shows the method of turnbuckle safetying used on the C-5A control cable systems.

Discussion

The requirement to minimize the number of cable interconnections is good, but compliance can be a matter of opinions which may differ. The other detail hardware requirements are good ones which can be practically demonstrated.

The "Discussion" section under Paragraph 3.2.3.2.4.3 of the "Users' Guide" AFFDL-TR-74-116 presents a good example of the "depth" of guideline information which should be reflected in all the "Discussion" sections of the "Users' Guide." This requirement has been satisfied by the C-5A design and should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.3.2.4.4 Cable Routing. Control cables shall be arranged in parallel runs and be accessible to inspection for their entire length. Cable runs located in aeroelastic structure, such as aircraft wings, shall be routed so as to minimize any induced control action caused by structural flexure. Spacing between adjacent cables shall prevent cables, turnbuckles, and fittings from chafing during all operating conditions including vibration. Slack return cables shall not snag on each other or any other equipment or structure when the controlling cables are loaded to design limit loads at the adverse extremes of temperature, structural deflection, and other operating conditions. Cables shall not be subjected to critical bends at the junction with cable terminals or other attaching points such as on drums and sectors.

Comparison

The C-5A mechanical flight control cable systems as shown in Figures 1 and 2 (3.2.3.1.1) are arranged in parallel runs which are accessible for inspection for the entire length of the run. A minimum number of pulleys and brackets were used since most of the cable runs were straight with a small number of changes in cable direction. For unsupported (by pulley) cable runs, low friction fairlead rollers (idlers) as shown in Figure 1 (3.2.3.2.4.4) are provided at least every 80 inches to minimize cable sag and assure clearance of adjacent structure during various modes of structural deflection. The cable runs located in structure subject to significant aeroelastic deflection, such as the aircraft's wings, have been routed as close as practical to the neutral bending axis in order to minimize the effect of induced control action. Fairlead rollers (idlers) were used to maintain the wing cable runs as close as possible to the neutral axis during these structural flexing modes. Adjacent cable assemblies are spaced to prevent cables, turnbuckles, and fittings from chafing or hanging up during all of the environmental operating conditions.

When primary control cables are loaded to design limit load, the slack return cable is prevented from any appreciable slackness by maintaining a cable pre-load from the "slack take-up" quadrant of the tension regulator. Adequate cable guards and fairlead rollers retain the cables on the pulley quadrants, etc., under all environmental operating conditions. The cable system is designed to provide generous or no bends at all junction points with cable terminals, quadrants, sectors, etc. All cable installations are routed such that they do not interfere with, block, or rub on any hinged cover or moveable equipment.

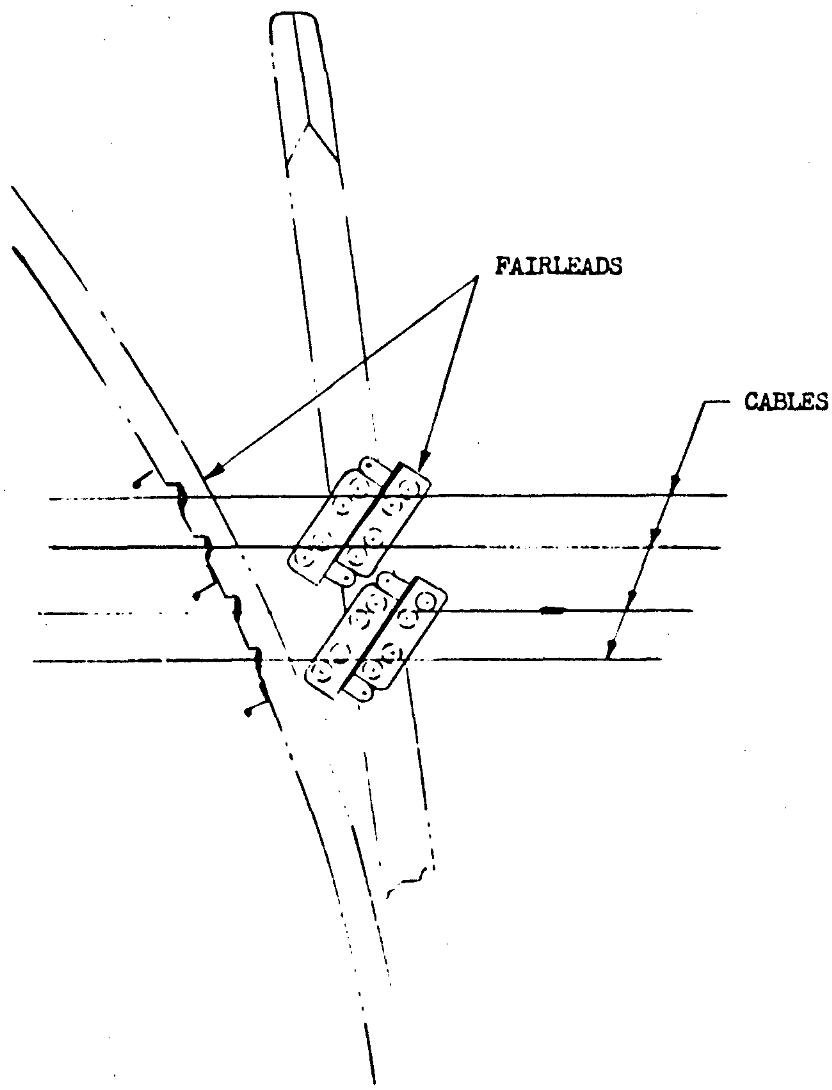


FIGURE NO. 1(3.2.3.2.4.4) C-5A CONTROL CABLE FAIRLEADS

Discussion

Meeting this type of requirement can be demonstrated during testing. This design requirement is valid and has been satisfied by the C-5A design and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.3.2.4.5 Cable sheaves. Cable drums, sectors, and pulleys of adequate capacity and diameter for their function and to meet aircraft life requirements shall be provided. They shall be large enough for the cable wrap angle such that the cable strands are not overstressed. The diameter and number of grooves on cable drums, and the radius and angle of control cable sectors shall be adequate for the required cable travel. Overtravel allowance shall not be less than 5 percent of full travel in either direction and at least 10 degrees. When cable wrap varies with cable travel, the initial wrap with the sheave in the neutral position shall be at least 115 percent of the full cable travel in either direction. If overtravel exceeds the minimum required, cable wrap shall be increased a corresponding amount. All cable grooves on drums and sectors, machined or die cast, shall have root radii properly sized for the cable size used thereon. Specific approval shall be obtained before using plain pulleys in essential applications. Antifriction pulleys used in flight control systems shall be MS standards in accordance with MIL-P-7034, and the design limit load shall not exceed the allowable limit load specified for the applicable standard.

Comparison

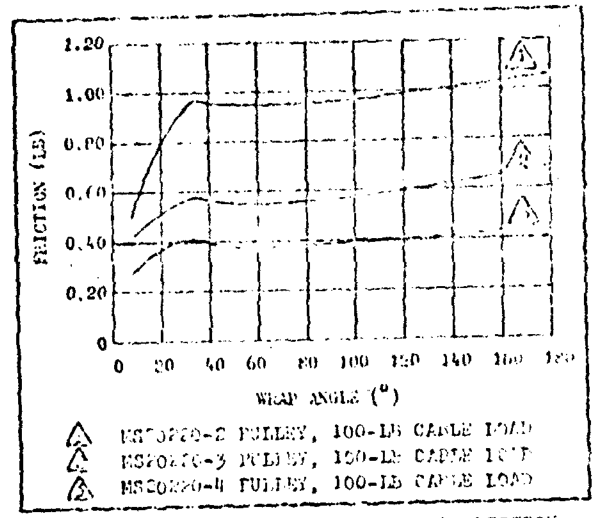
The C5A flight control cable systems cable drums, pulleys, sectors, and quadrants, etc. were designed to meet or exceed the system functional and endurance life requirements. Detail system design criteria included considerations of cable friction, strength, component wear, etc.

Since cable friction forces vary inversely with pulley size the largest practical pulley and quadrant diameters were used. This relates to cable friction increasing sharply with the amount of cable wrap up to one pitch length as noted on Table 1 of Figure 1 (3.2.3.2.4.5). Therefore where a significant change in cable direction was required a single pulley of higher wrap angle (up to 90°) was used (instead of using several pulleys having smaller amounts of cable wrap) resulting in less friction and a lighter system. The larger quadrant and pulley diameters also reduced the stress levels of the cable strands.

The design of all the flight control cable systems provided for cable overtravels which exceeded ± 5 percent of the maximum normal system travel and more than ± 10 degrees of overtravel rotation on all of the quadrants and sectors.

All of the machined or die cast cable quadrant grooves are designed to provide the proper root radii in accordance with the applicable military specification groove size requirements. Anti-friction pulleys were selected to MS standards in accordance with MIL-P-7034.

All of the cable system quadrants, sectors, pulleys, etc. were designed or selected such that the design limit load did not exceed the allowable limit load as specified by the applicable specifications.



EFFECT OF PULLEY SIZE ON FRICTION - 1/8-INCH ZINC-COATED CABLE

TABLE I

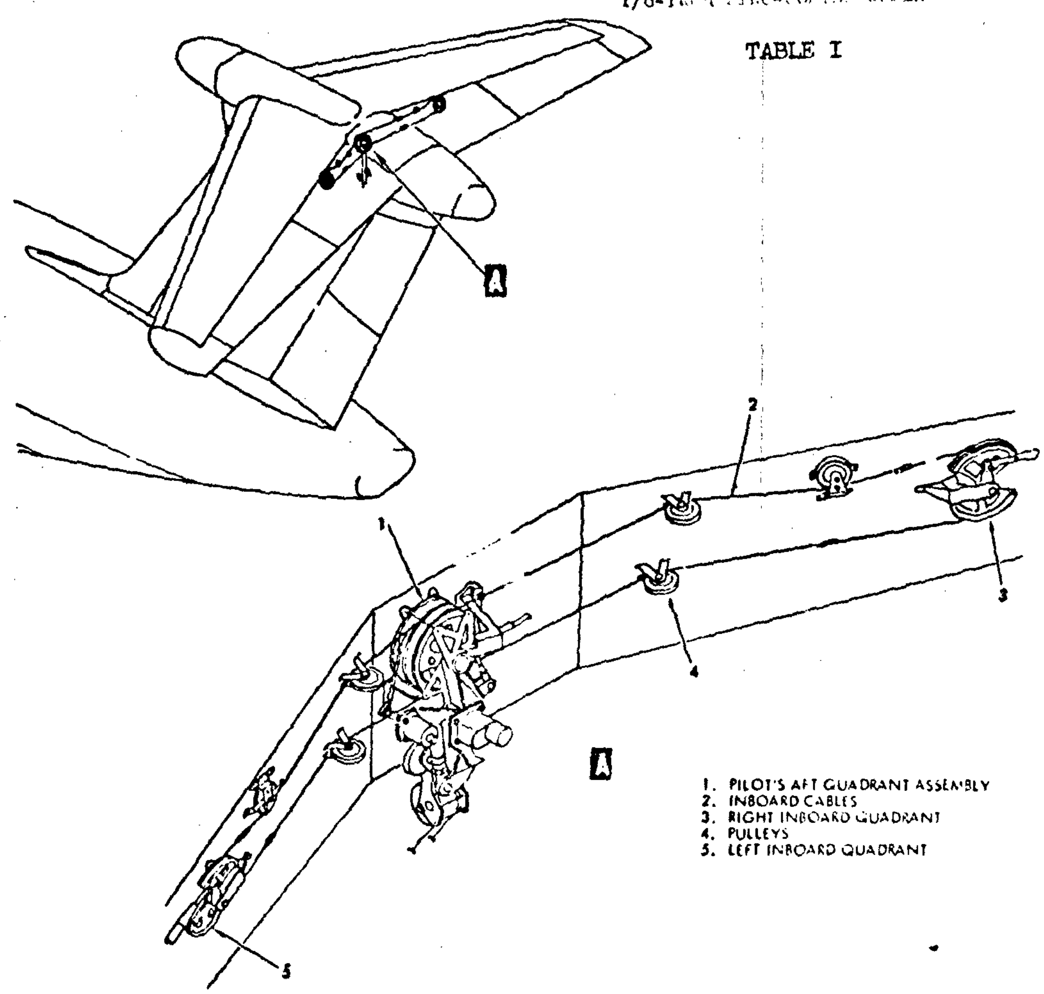


FIGURE 1 (3.2.3.2.4.5) C-5A CABLES AND SHEAVES (TYPICAL)

Discussion

This is a good requirement which has been satisfied by the C5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the Specification "as is".

Requirement

3.2.3.2.4.6 Cable and Pulley Alignment. Fixed-mounted pulleys shall be aligned with their cables within 2 degrees as specified in AFSC Design Handbook DH 2-1, DN 3B1, Subnote 1.1.3(1), Cable Pull. Where a control cable has an angular motion with respect to the plane of the pulleys, the maximum misalignment resulting from this motion must not exceed 2 degrees, and the cable shall not contact the pulley (or quadrant) flange for the total cable travel.

Comparison

The C-5A flight control cable systems were generally designed for 0 degrees nominal misalignment between the centerline of the cable and the centerline of the plane of the pulley, quadrant, sector, etc. In no case was this nominal design misalginment allowed to exceed 0.5 degrees. Considering tolerance effects, the maximum misalignment was therefore less than the 2 degree limit. This requirement also applied anytime a control cable had an angular motion with respect to a pulley, quadrant, etc.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.3.2.4.7 Pulley-Bracket Spacers. Loose spacers between pulleys, bearings, and pulley brackets shall not be used.

Comparison

The C-5A flight control system uses loose spacers on most of the pulleys, pulley bracket, and bellcrank applications. On applications where a "gang" of pulleys are used on a single mounting bracket the use of spacers was the most economical method of spacing the pulleys. If the definition of this requirement includes spacer bushings, then the use of a sliding bushing on one side of a pulley bearing or bellcrank bearing, as shown in Figure 1 (3.2.3.2.4.7) is the most practical method of "clamping up" on the bearing without prestressing the legs of the mounting bracket or clevis joint. On this type of application, a flanged bushing may be pressed into the mounting bracket on the other side of the bearing being installed. Without the use of the sliding bushing or spacer the gap between the bearing mounting faces would have to be controlled by a[illegible] onal mounting flanges or closer tolerances. The thickness [illegible] mounting flanges would have to be designed to accommodate a "clamp up" residual stress factor. These alternatives would result in a more expensive and heavier design.

Discussion

This requirement was not met by the C-5A design. It is felt that this requirement is not valid and can result in increased cost since it allows [illegible] r no other judgements. The objective of minimizing the number of loose components or components which can be incorrectly installed is a good one, but should be evaluated as a cost effective and weight criteria. Therefore, this requirement would be good as a design goal for future transport aircraft.

Recommendation

Revise the wording of specification 3.2.3.2.4.7 to read:

> "The use of loose spacers between pulleys, bearings, and pulley brackets shall be avoided except when their use can be shown to provide a cost effective solution without undue maintenance problems."

Additional Data (For Users Guide)

The use of loose spacers should be held to a minimum commensurate with cost effective design. The use of sliding spacers or bushings can be an effective method of 'clamping up' on the pulley or bellcrank bearing without prestressing the legs of mounting brackets or clevis joints. Reference Figure [No. 1(3.2.3.2.4.7)] represent a typical bearing clamp up using sliding bushings.

NOTE: For Figure [ ] use next available users guide number.

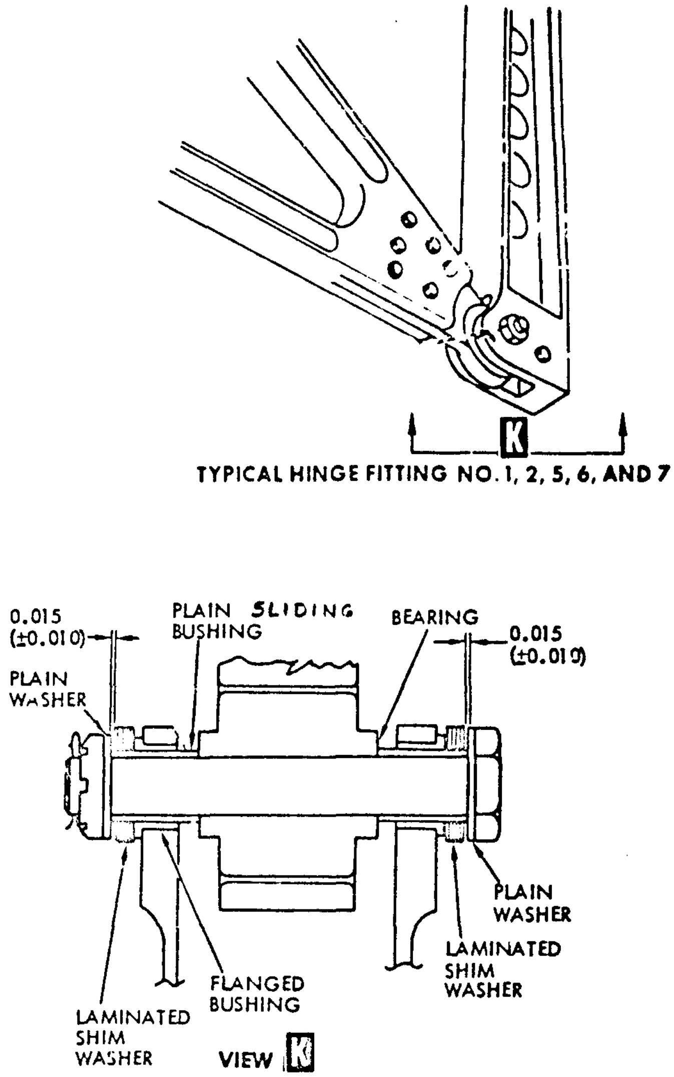


FIGURE NO. 1(3.2.3.2.4.7) C-5A TYPICAL BEARING CLAMP UP

Requirement

3.2.3.2.4.8 Sheave Guards. Guards shall be installed at all sheaves (pulleys, sectors, drums, etc.) as necessary to prevent the cable from jumping out of the groove of the sheave. Guards shall be installed at the approximate point of tangency of the cable to the sheave. Where the cable wrap exceeds 90 degrees, one or more intermediate guards shall be installed. All guards shall be supported in a way which precludes binding of the sheave due to relative deflections in the aircraft structure. Additional guards shall be installed on sectors as necessary to ensure retention of the cable end in its attachment under slack cable conditions. The design of the rubbing edges of the guard and the selection of materials shall be such as to minimize cable wear and prevent jamming even when the cable is slack.

Comparison

The C-5A flight control cable systems used cable guard pins on all pulleys, sectors, drums, etc., to prevent cables from jumping out of guide grooves. The guard pins were located at the approximate tangency point of the cable wrap angle. The guard pins were generally mounted on the same structural member as the pulley or quadrant to preclude the possibility of binding due to relative deflections from aircraft structure.

More than one guard pin was sometimes used for greater cable angle wraps and additional guard devices (safety wire and cotter pins) were used on sectors where a slack cable might permit disconnect of the cable end from the sector and cause binding or loss of cable function. Figure 1 (3.2.3.2.4.8) shows a guide for the application of the guard pins which was used for the C-5A design. Table 1 indicates the number of guard pins required as a function of the cable wrap angle. The cable guards used were round and the materials such that in the event of a slack cable the cable wear was minimized and the chances of a jam were precluded.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Additional Data (For Users Guide)

The table shown in Figure [No. 1(3.2.3.2.4.8)] is a representative industry guideline for guard pin locations.

NOTE: For Figure [ ] use next available users guide number.

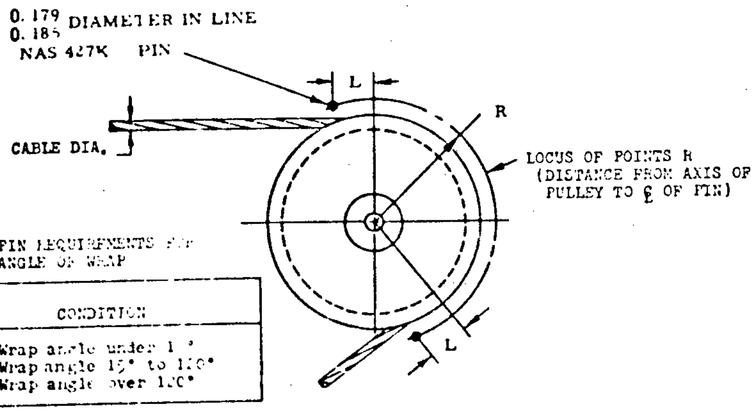


TABLE I - GUARD PIN REQUIREMENTS FOR GIVEN ANGLE OF WRAP

MIN. NO. OF GUARD PINS	CONDITION
1	Wrap angle under 1[illegible]°
2	Wrap angle 15° to 1[illegible]0°
3	Wrap angle over 1[illegible]0°

CALCULATED GUARD PIN LOCATIONS

PULLEY CLEARANCE	CABLE CLEARANCE	PIN	PULLEY	CABLE DIA.	R	L (MAX.)
1/32	1/16	NAS427K Guard Pin	MS20219(-,A)2, NAS376-2FP, or LS9019A2	1/16	1.004	0.334
			MS20219(-,A)3, NAS376-3FP, or LS9019A3	3/32	1.004	0.214
			MS20219(-,A)4, NAS376-4FP, or LS9019A4	1/16	1.431	0.387
			MS20219(-,A)5, NAS376-5FP, or LS9019A5	3/32	1.431	0.242
			MS20220(-,A)1 or LS9020A1	1/8	1.004	0.3[illegible]
				5/32	1.004	0.316
				3/16	1.004	0.195
			MS20220(-,A)2 or LS9020A2	1/8	1.629	0.504
				5/32	1.629	0.358
				3/16	1.629	0.220
			MS20220(-,A)3 or LS9020A3	1/8	2.254	0.595
				5/32	2.254	0.453
				3/16	2.254	0.262
			MS20220(-,A)4 or LS9020A4	1/8	2.879	0.674
				5/32	2.879	0.518
				3/16	2.879	0.2[illegible]
			MS20221(-,A)1 or LS9021A1	3/16	1.441	0.465
				7/32	1.441	0.353
				1/4	1.441	0.196
			MS20221(-,A)2 or LS9021A2	3/16	2.191	0.576
				7/32	2.191	0.437
				1/4	2.191	0.2[illegible]
			MS20221(-,A)3 or LS9021A3	3/16	2.941	[illegible]
				7/32	2.941	0.507
				1/4	2.941	0.27[illegible]
			MS[illegible]A1	1/16	0.770	[illegible]
				3/32	0.770	0.172

FIGURE NO. 1(3.2.3.2.4.8) C-5A GUARD PIN LOCATIONS

Requirement

3.2.3.2.4.9 Sheave Spacing. In any given cable run, no portion of the cable shall ever pass over more than one sheave.

Comparison

The C-5A flight control cable system design provided a spacing greater than the maximum cable travel between all cable pulleys, quadrants, sectors, etc., on the same cable element. This consideration provided for reduced wear, increased cable endurance life, and reduced friction. Not only does this preclude the transferral of cable twist from sector to sector resulting in increased wear and friction, but it also prevents stress reversals of the cable fibers to give better fatigue life.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.3.2.4.10 Cable Tension. Cable rig loads shall insure positive cable tension in control and return legs of closed-loop cable installations under all operating conditions including airframe deflection and differential expansion and contraction between the cable and airframe structure throughout the designed operating temperature range. The cable return leg may be allowed to go slack when the control leg is loaded above the normal operating load, providing it cannot snag, when the control leg is loaded at any load up to limit load, and that there is no hazardous loss of system performance. Cable tension regulators shall be provided only if positive cable tension cannot be maintained in both legs, with reasonable rigging loads.

Comparison

The C-5A flight control cable system was designed to provide the required cable tension in the closed-loop cable installations under all operational and environmental conditions.

On the long cable runs, as shown in Figure 1 (3.2.3.2.4.10), tension regulators are used to maintain the cable tension rig load. On shorter cable runs, as shown in Figure 2 (3.2.3.2.4.10), the cables are rigged to a load which meets all of the load and environmental operating conditions while satisfying the functional requirements of input response and operating loads. The cable rig loads were selected to provide positive tension under all combinations of temperature changes, structural deflection, and applied load to prevent cables from becoming slack and sagging into adjacent equipment. Since the cable system friction is directly proportional to the cable load the cable rig tension was held as low as possible.

Factors which determined the rig tension requirements are listed below in two categories.

Thermal Effects

- Differences in expansion and contraction between the cable and airframe over the full range of operating temperatures
- Heating from adjacent equipment
- Effects of temperature lead or lag on cables and airframe resulting from rapid temperature change

Mechanical Effects

- Effects of aircraft structure deflections on cable load and movement resulting from maneuver loads or equipment loads.

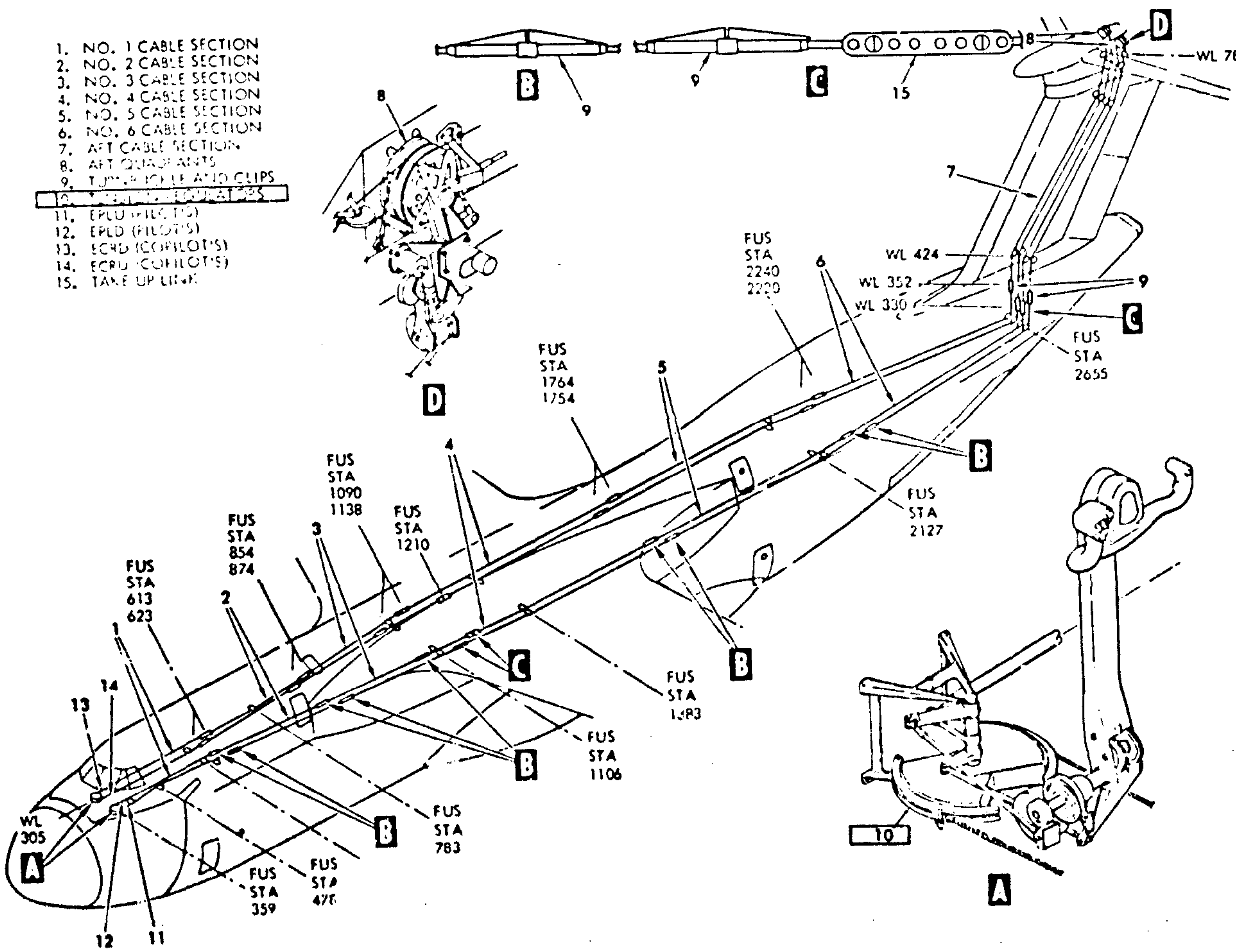
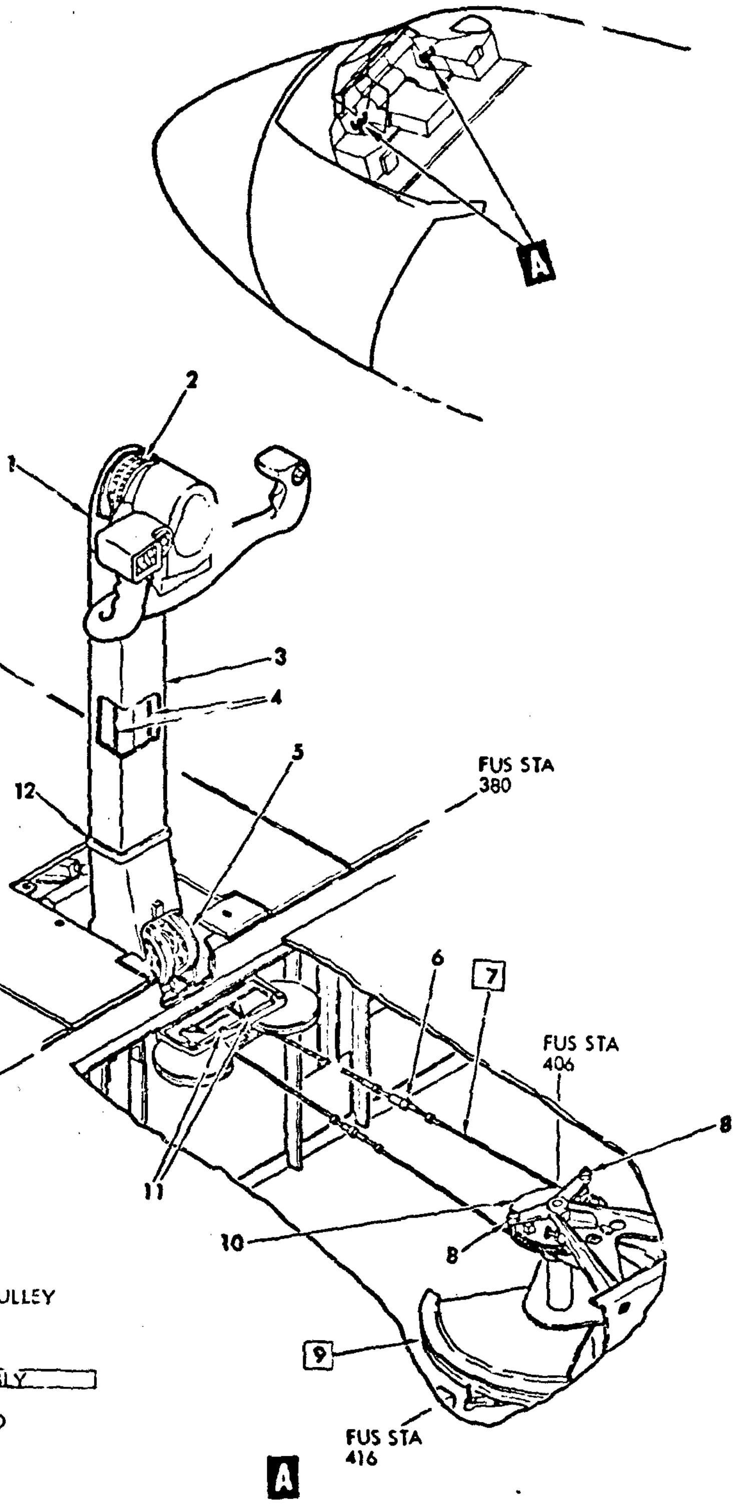


FIGURE 1 (3.2.3.2.4.10). C-5A ELEVATOR CONTROL CABLES



1. COVER
2. UPPER CONTROL COLUMN CABLE PULLEY
3. CONTROL COLUMN
4. CONTROL COLUMN CABLE ASSEMBLY
5. LOWER CONTROL COLUMN PULLEY
6. TURNBUCKLE
7. CABLE ASSEMBLY
8. GUARD PIN ASSEMBLIES
9. TENSION REGULATOR ASSEMBLY
10. QUADRANT
11. CENTER GUIDE PULLEY GUARD PIN ASSEMBLIES
12. BOOT CLAMP

FIGURE 2 (3.2.3.2.4.10) C-5A CONTROL COLUMN CABLES

* Effects of structural movement on cables during pressurization cycles

* Breakout friction

The closed loop cable systems are designed so that they never become excessively slack under the most adverse operating condition. This minimizes the possibility of a loose cable from hanging up and prevents unnecessarily "spongy" cable system with its increased dead band and resultant loss of increment control and response.

Discussion

Cable tension regulators permit reduced cable tension with resultant reductions in system breakout and improved centering. The last sentence of the requirement therefore is too restrictive. It is suggested that more specific requirement guidelines be provided in the "Users' Guide." This is generally valid, has been satisfied by the C-5A design and should be specified for future transport type aircraft.

Recommendation

Revise the requirement as follows:

Delete the last sentence

Requirement

3.2.3.2.4.11 Cable Tension Regulators. When used, tension regulators shall maintain required tension at all times. Integral calibration shall be provided to show proper cable tension without the use of external tensionmeters or other equipment.

Comparison

Tension regulators were required for the long lengths of the mechanical input cable installation for the C-5A flight control system as shown in Figure 1 (3.2.3.2.4.10). The tension regulators insured positive cable tension in the control and return legs of the closed loop cable installations during all the operational, environmental, and limit load operating conditions. The initial design requirement was based on a trade-off study of the functional requirements and penalties of the use of tension regulators versus a "soft" cable system with no tension regulators, considering the parameters of weight, complexity, cost, and failure modes. The tension regulators also provided slack take-up to prevent excessive sagging of the return cable leg during a high or limit load operating condition. The type of tension regulator used in the C-5A control system is shown in Figure 1 (3.2.3.2.4.11). Spring loaded sectors and a sliding "crosshead" member maintained the required cable tension. A tension regulator scale as shown in Figure 1 (3.2.3.2.4.11) B shows the cable tension.

Integral surge locks are incorporated into the tension regulators which automatically "lock up" the two quadrant sectors in the event of a broken cable, thus preventing a "hard over" signal from being generated by the free sectors' compensation spring.

Discussion

The C-5A flight control system was designed to use cable tension regulators that meet this requirement.

This is a good requirement which can be practically demonstrated and should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

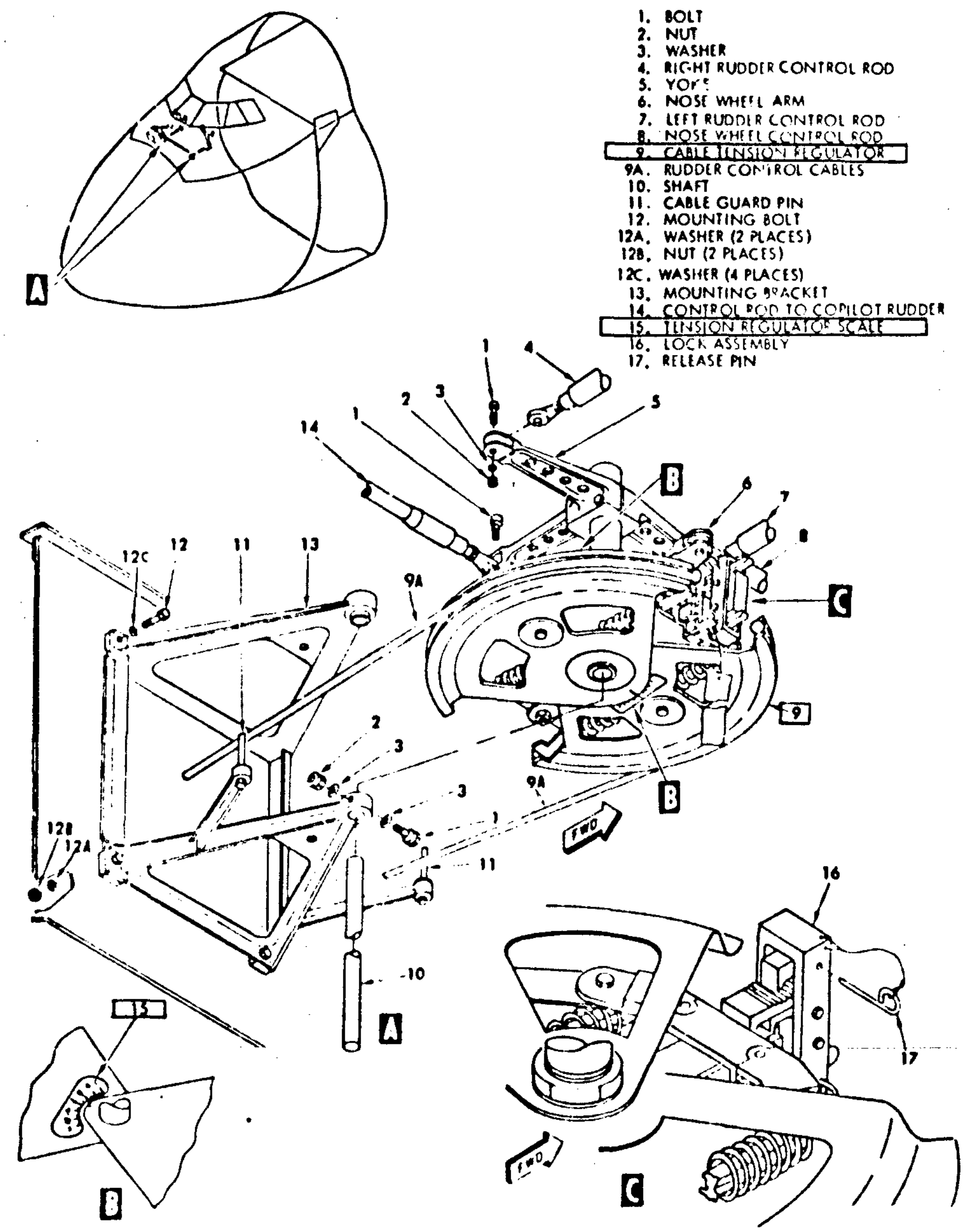


FIGURE NO. 1(3.2.3.2.4.11) C-5A RUDDER CONTROL CABLE TENSION REGULATOR

Requirement

3.2.3.2.4.12 Fairleads and Rubbing Strips. Fairleads shall not cause any angular change greater than 3 degrees in the direction of the cable under all conditions including those due to structural deflections in flight. Fairleads shall be split to permit easy removal unless the size of the hole is sufficient to permit the cable with swage terminals to be threaded through.

Comparison

The C-5A flight control cable systems do not use any rigidly mounted fairleads or rubbing strips. To keep cable friction to a minimum, fairlead rollers (cable idlers) were used to keep cables from chafing, snagging and slapping against each other or adjacent parts of the aircraft. Even with fairlead rollers the angular change of cable direction was kept to very small angles. The pressure seals are a non-metallic material which the cable runs through, but this seal is allowed to ride in an oversize hole such that no angular change of direction of the cable occurs. These pressure seals are split to permit their easy removal.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Additional Data (For Users Guide)

An alternative to fairlead rub strips are fairlead rollers such as those shown in Figure [No. 1(3.2.3.2.4.4)]. The fairlead rollers result in less friction than the rub strips.

NOTE: For Figure [ ] use the next available users guide number.

Requirement

3.2.3.2.4.13 Pressure Seals. Pressure seals shall meet compartment sealing requirements within cable installation friction requirements. They shall be designed to preclude jamming the control system.

Comparison

The C-5A flight control cable system uses a pressure seal like the one shown in Figure 1(3.2.3.2.4.13) to meet the specified pressure sealing requirements of a pressurized compartment and maintain a minimum of cable friction. The "seal" on this installation, as on any low friction seal, is in fact a controlled leakage orifice which meets the allowable compartment leakage requirements. The inner "pressure seal set" may be installed or replaced after the cables are installed and this "seal set" also accommodates cable misalignment without increasing the cable friction.

The seal material is such that if a jam should occur, although this is very improbable, the softer flanges of the "pressure seal set" could be sheared out by a higher pilot force to clear the jam.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Additional Data (For Users Guide)

Pressure seals should be designed to minimize misalignment of the cable from structural deflection or production tolerances in order to minimize cable friction. One method of accomplishing this is to provide some type of 'slotting' which will allow the 'seal set' to move with the cable misalignment and which still maintains its sealing qualities. Figure No. [1(3.2.3.2.4.13)] represents Lockheed's particular patented design solution.

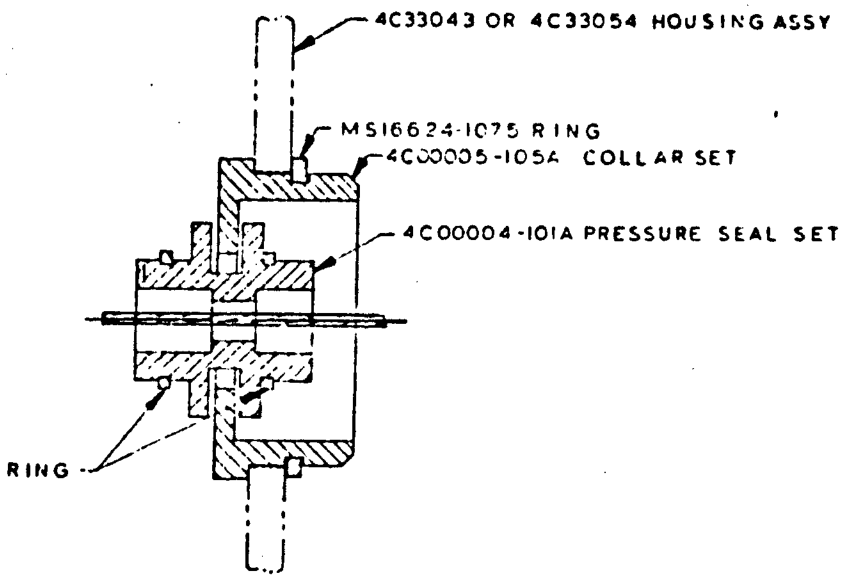


FIGURE NO. 1(3.2.3.2.4.13) C-5A CABLE PRESSURE SEAL

Requirement

3.2.3.2.5 Push-Pull Rod Installations. Push-pull rod installations shall be designed to preclude binding or separating from the mating linkage and shall permit servicing and rigging.

Comparison

The C-5A flight control system (FCS) used push-pull rod installations only in selected locations within the control cable system installations for certain specialized functions. Figure 1 (3.2.3.2.5) illustrates a typical push-pull rod installation which is used as an interconnect between two independent cable systems using a rod which incorporates a shear out capability. Figure 2 (3.2.3.2.5) illustrates the use of an override bungee and trim actuator installed in series with the input system and serving a dual function of providing the normal pilot input push-pull rod motion in addition to their alternate functions. All push-pull rods and equivalent components were designed to preclude binding and separation from the mating linkage. The rod end bearing on one end of each rod is threaded in, then fixed by a rivet. The bearing rod end at the other end of the rod assembly is adjustable, but can be locked in position after adjustment by the use of a jam nut in combination with either an NAS513 or NAS1193 lock washer as shown in Figure 3 (3.2.3.2.5). All the push-rod attachments are bolted to the clevis joint by a close tolerance bolt and a self-locking castellated nut which is cotter pinned.

All push-pull rod installations are designed and located to permit easy servicing and rigging. Where it was necessary to assure easy servicing and rigging, access doors or plates were included in the design.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

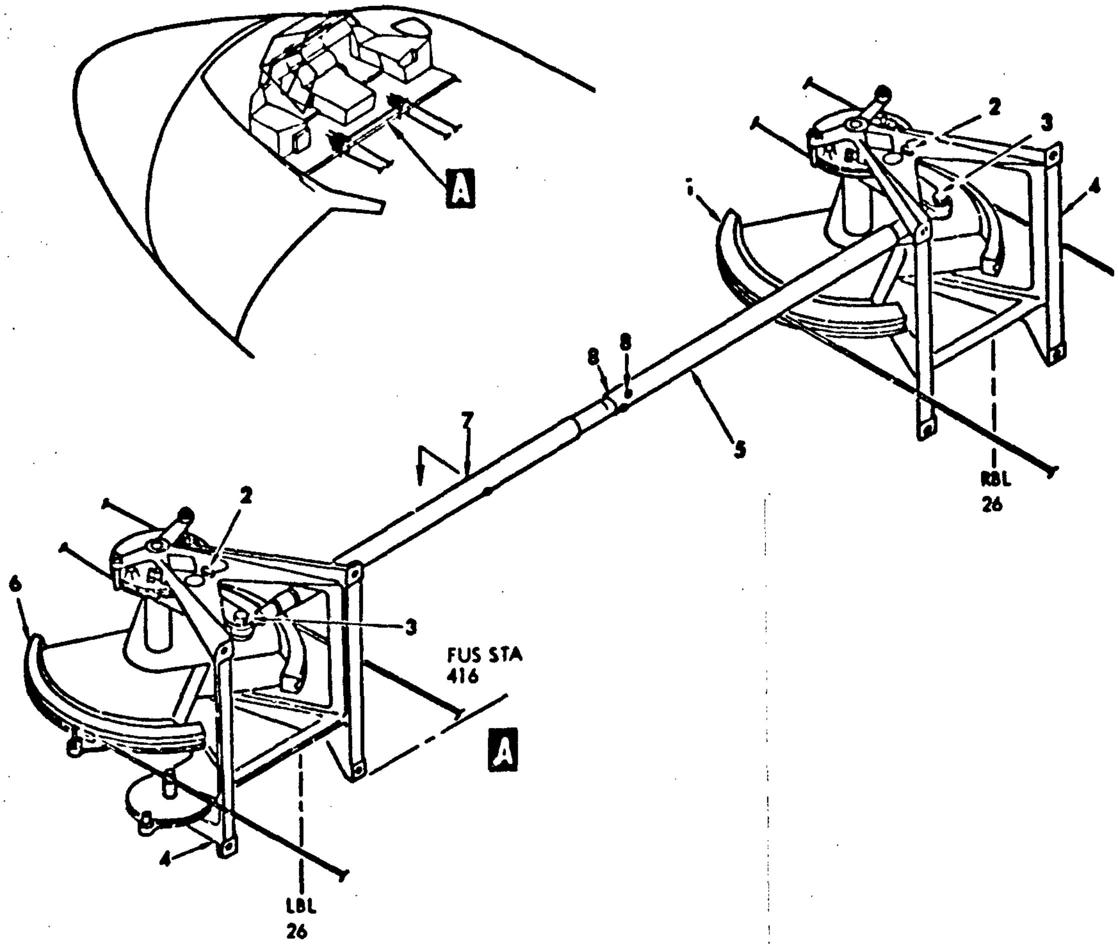
Recommendation

Accept the specification "as is."

Additional Data

Figure [No. 1 (3.2.3.2.5)] illustrates a typical push-pull rod installation, in this case for interconnecting redundant cable systems. Sometimes, one end of a rod is a bearing rod end and the other is a fork end. Other rods may use either a bearing rod end or a fork end on both ends of the rod. In most applications, only one end is free to be adjusted and the other end is fixed similar to the arrangement shown in Figure [No. 3 (3.2.3.2.5)]. In a few applications, both ends of the rod are made adjustable for ease of maintenance. These designs permit rod length adjustment in one-half turn increments of the rod end only. In rare instances, where finer adjustments to rod length are required, a turnbuckle arrangement is used employing left-hand threads on one end and right-hand threads on the other end. In all cases the rod ends are either permanently fixed to the rod or mechanically safetied against rotation. Typical safetying includes slotted rod and fittings with NAS513 or NAS1193 washers, jam nuts and safety wire.

Note: For Figure No. [ ] use next available "Users' Guide" number.



1. COPILOT'S TENSION REGULATOR ASSEMBLY
2. RIG PIN HOLE
3. BOLT ASSEMBLY (BOLT, WASHER, NUT, COTTER PIN)
4. BRACKET ASSEMBLY
5. INTERCONNECT ROD
6. PILOT'S TENSION REGULATOR ASSEMBLY
7. SHEAR RIVET
8. SCREW ASSEMBLY (SCREW, WASHER, NUT, COTTER PIN)
9. OUTER TUBE
10. CENTER TUBE SECTION
11. FILLER

FIGURE NO. 1 (3.2.3.2.5) AILERON TENSION REGULATOR INTERCONNECT ROD

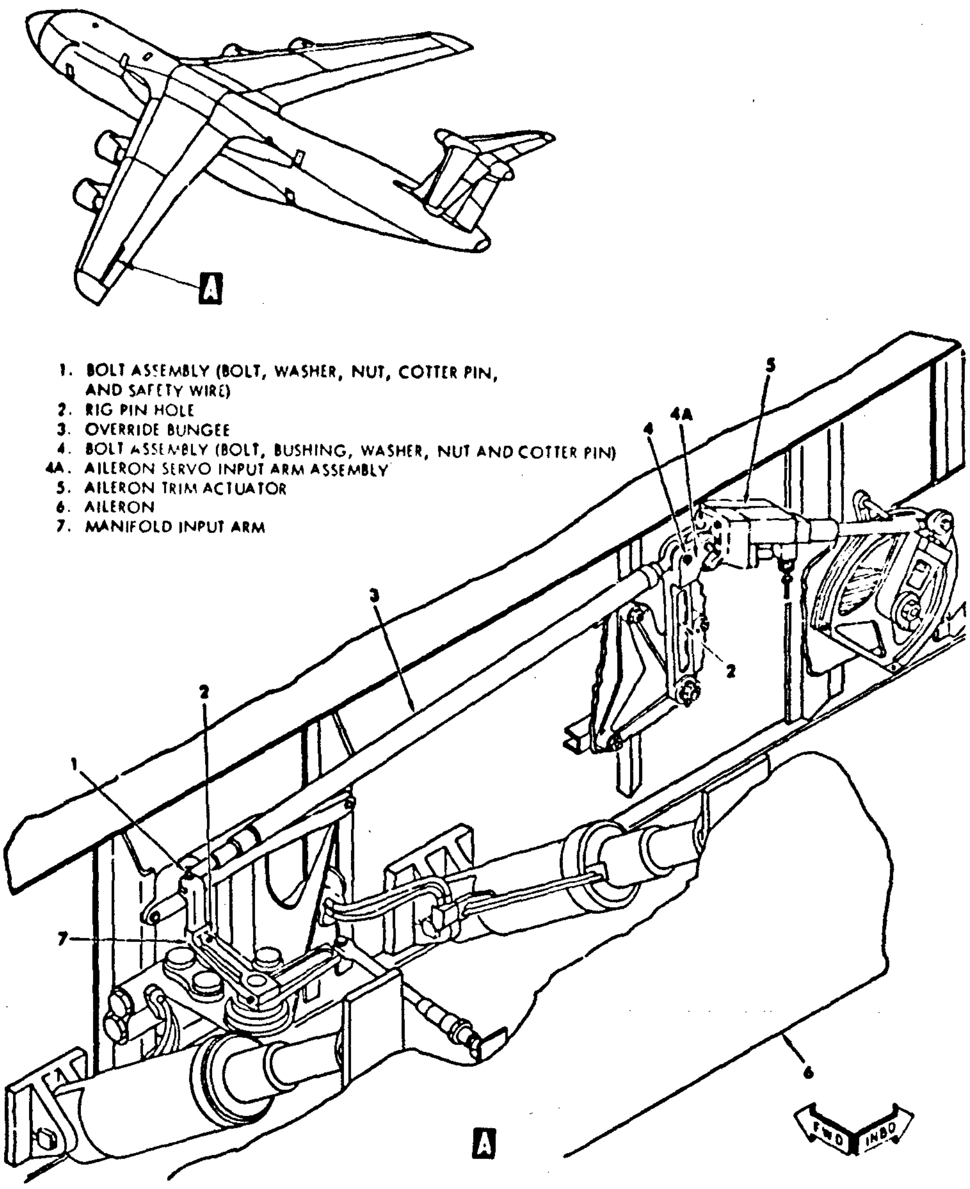
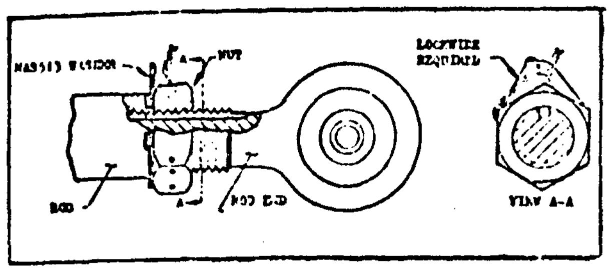
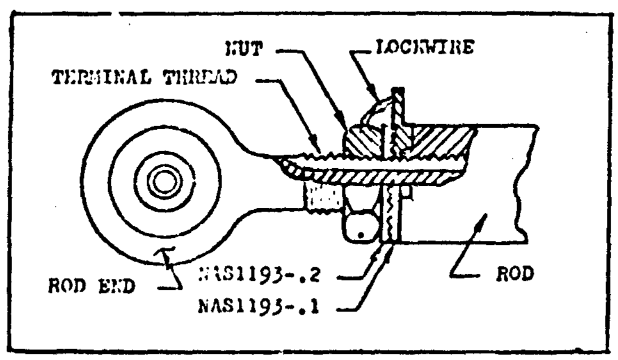


FIGURE 2 (3.2.3.2.5) AILERON SERVO INPUT SYSTEM



TYPICAL INSTALLATION OF
NAS513 LOCKING DEVICE



TYPICAL INSTALLATION OF
NAS1193 LOCKING DEVICE

FIGURE NO. 3 (3.2.3.2.5) TYPICAL ROD END LOCKING DEVICES

Requirement

3.2.3.2.5.1 Push-Pull Rod Assemblies. Push-pull rod assemblies shall be designed and installed such that inadvertent detachment of adjustable terminals is impossible, and such that any change in length due to loosening of the terminals cannot result in an unsafe condition. On any single rod assembly, adjustment shall be possible at one end only. The fixed end of each rod shall be attached to its mating linkage element in a manner which precludes rotation of the installed assembly. The adjustable end shall be of the clevis type or join a clevis type in such a manner that it is also prevented from rotating. When an unsymmetrical rod is used, such as one with a cutaway portion to allow for relative motion of an attached link, the rod end terminals and mating linkage elements shall positively prevent incorrect installation of the rod. Push-pull rods shall have a minimum wall thickness of 0.035 inches and shall be capable of withstanding loads of 1.5 times limit loads in both tension and compression without failure, buckling, or any other form of permanent deformation. All joints shall be made in a manner which precludes loosening and fatigue failure. All closed cavities in rod assemblies installed in unpressurized spaces shall be provided with drain holes adequate to drain ingested water unless cavities are air tight. All push-pull rod terminals shall incorporate antifriction bearings as specified in 3.2.7.2.1.1 or self-lubricating spherical bearings as specified in 3.2.7.2.1.2. All terminal pins shall be retained as specified in 3.2.8.3.2.?. Loose washers or other loose spacers shall not be used to maintain terminal spacing in the connecting linkage.

Comparison

The C-5A flight control system push-pull rod assemblies are designed and installed to prevent inadvertent detachment or change in length due to loosening of the adjustable rod end attachments thereby precluding development of an unsafe condition. Push-pull rod assemblies or components are usually designed to have adjustment at one end only. The fixed end of the push-pull rod assembly is always installed in a manner which precludes rod assembly rotation. Figure No. 1 (3.2.3.2.5.1) illustrates the usual manner of push-pull rod installation which has the rod end bearing installed between the bellcrank clevis lugs thus preventing rod rotation. The adjustable rod end uses a mechanical locking device such as an NAS1193 or NAS513 lock washer, jam nut and safety wire as shown in Figure No. 3 (3.2.3.2.5).

The push-pull rod end bearings are either an antifriction bearing as specified in 3.2.7.2.1.1 or self-lubricating as specified in 3.2.7.2.1.2.

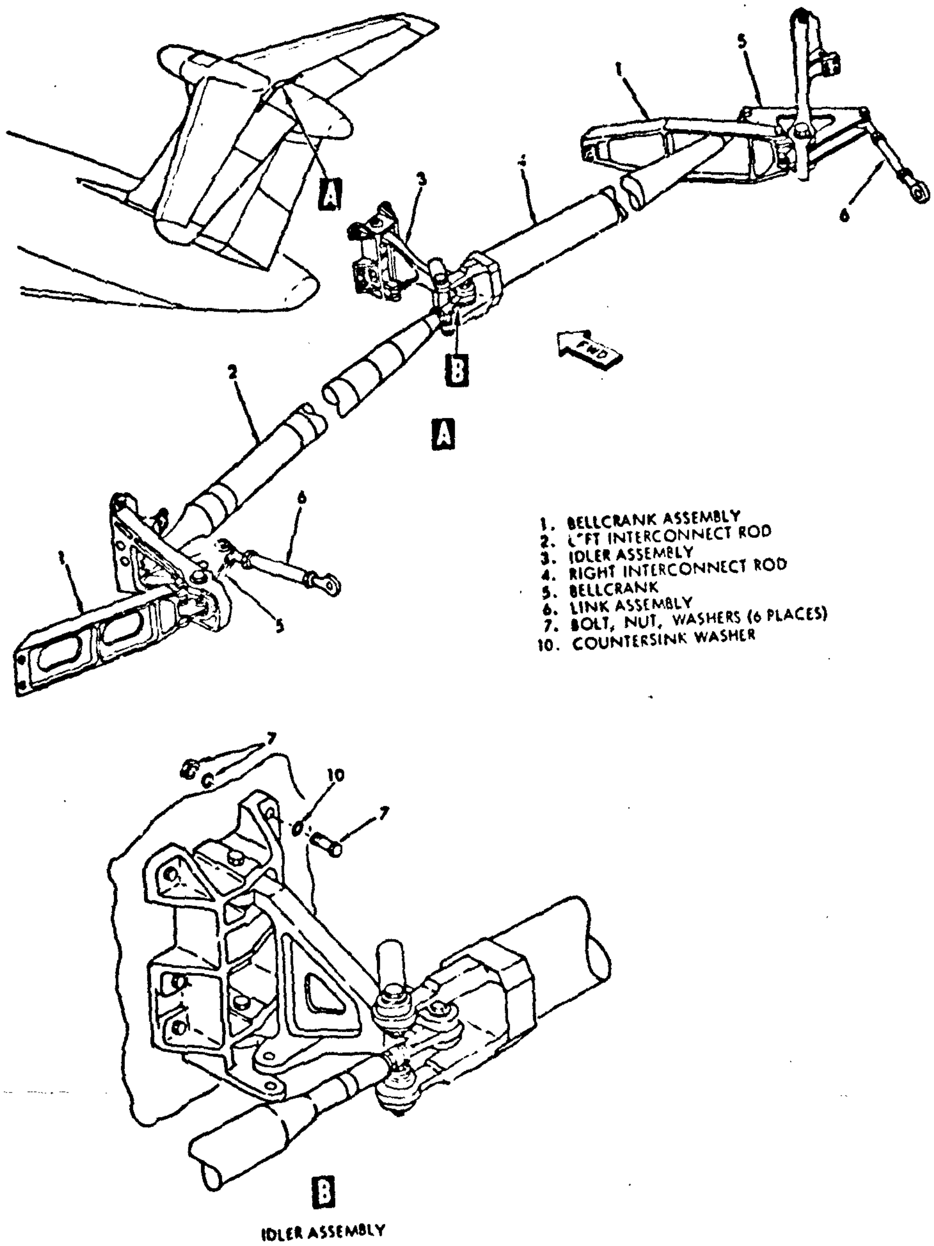


FIGURE 1 (3.2.3.2.5.1) INBOARD ELEVATOR INTERCONNECT PUSH-PULL ROD INSTALLATION

All push-pull rods and components were designed to meet the limit load criteria of 3.2.3.2.1 which was much higher than the normal maximum operating loads. However, the ultimate load criteria was 1.5 times limit load whereby component deformation was allowed.

Discussion

The detail design criteria required by this specification is good, except for two areas concerning the load requirement and the use of loose spacers or bushings which were not met by the C-5A design. The load requirement that the push-pull rods shall be "capable of withstanding loads of 1.5 times limit load without failure, buckling, or any other form of permanent deformation" is not consistent with the general stress criteria specified in MIL-Handbook 5 and the industry standards. Normally a factor of 1.5 times limit load constitutes an ultimate load criteria which does allow deformation of the member. The other area of concern is the requirement to prohibit the use of any loose spacers (or bushings) to maintain spacing in the connecting linkage joint.

A change has been recommended to change paragraph 3.2.3.2.4.7 to specify minimizing the use of loose spacer or bushings as a design goal. As shown in Figure No. 1 (3.2.3.2.4.7) loose sliding bushings are used to maintain a tight joint against the rod end attach bearing within the clevis joint. This has been done in the C-5A FCS in order to allow a greater (more economical) tolerance between the clevis faces in addition to precluding the effects of stress corrosion from clamp up prestressing on the clevis faces.

The specification as recommended for modification is good. It has been met by the C-5A design and can be readily demonstrated. It is recommended that this requirement be specified for all future transport type aircraft.

Recommendation

Revise the fifth sentence of 3.2.3.2.5.1 to read:

> "Push-pull rods shall have a minimum wall thickness of 0.035 inches and shall be capable of meeting the ultimate load factor of 1.5 times limit load in both the tension and compression directions."

Revise the last sentence to read:

> "The use of loose spacers to maintain terminal spacing in the connecting linkage shall be held to a minimum."

Requirement

3.2.3.2.5.2 Levers and Bellcranks. Applicable requirements in AFCS Design Handbook DH 1-6; System Safety, Section 3J; Flight Control Systems, Design Note 3J2; Mechanical Flight Controls; Pulleys, Brackets and Bellcranks, and Design Note 3JX; Safety Design Checklist, shall be met. Bearings shall have adequate self-aligning capability if necessary to prevent excessive deflection loads on levers and bellcranks, and, their installations shall be designed for easy replacement so that the parent part may be reused. Levers and bellcranks designed with dual load paths having the two sections positively joined by permanent fasteners, such as rivets, shall be bonded with adhesive.

Comparison

All levers and bellcranks used in the C-5A flight control systems have been designed to follow the same criteria as those defined for the mechanical flight controls in AFSC Handbook DH 1-6, Section 3J. The bellcrank and push-pull rod installations used self-aligning bearings to accommodate system misalignment from kinematics, installation, and tolerance effects. Bellcrank installations were located close to structural components to provide maximum protection and stiffer mounting. Applied reliability and maintainability requirements assured system simplicity and easy access to the components. Safety considerations were assured in the design process by rigid application of the reliability and safety requirements specified in the Contract End Item (CEI) specifications.

Discussion

This is a good requirement which has been satisfied by the C-5A control systems and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.3.2.5.3 Push-pull Rod Supports. Where long sections of push-pull rods are utilized in applications where jamming is not extremely remote, guides shall be installed at intervals to preclude fouling in the event of rod failure.

Comparison

The C-5A flight control system (FCS) used wire jumpers as push-pull rod guides for the aileron system interconnect as shown in Figure No. 1(3.2.3.2.5.). The jumpers would serve as a guide for the push-pull rod to preclude a possible jam condition in the event of a broken rod or loss of a joint connection. An extensive analysis of the FCS push-pull rod/bellcrank systems revealed this location to be the only one where a jam potential, following a failure, was possible.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design. It can be readily demonstrated and is valid for all future transport type aircraft.

Recommendation

Accept the specification "as is."

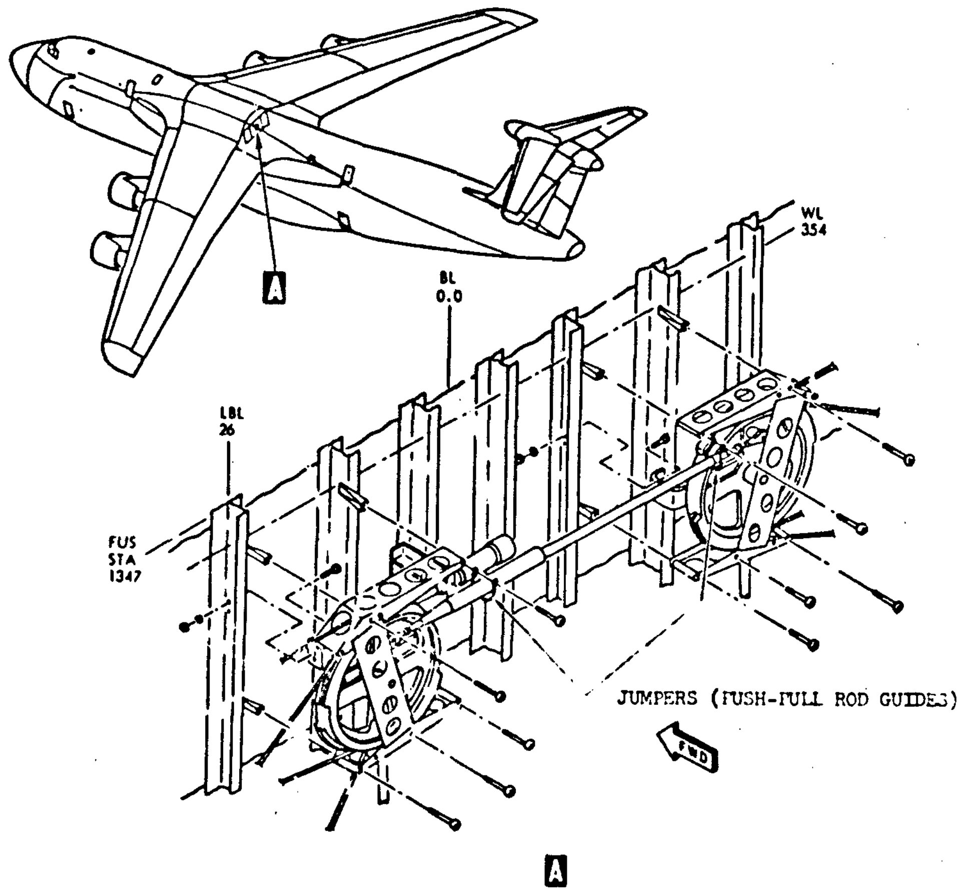


FIGURE 1 (3.2.3.2.5.3) AILERON QUADRANTS - REAR BEAM

Requirement

3.2.3.2.5.4 Push-pull Rod Clearance. Clearance between push-pull rods, and between rods and aircraft equipment and structure, shall be as specified to permit removal of adjacent LRU's without disconnecting the rods.

Comparison

The minimum clearance between the flight control system (FCS) push-pull rods, bellcranks, components, and structure have exceeded the requirements specified in 3.2.3.1.2 to insure that no probable combinations of temperature effects, air loads, structural deflections, vibrations, manufacturing tolerances or wear can cause binding or jamming. The FCS design has provided for easy accessibility and maintenance of the push-pull rod/bellcrank systems such that any adjacent components may be removed without disconnecting the rods. Typical FCS push-pull rod systems are shown in Figures No. 1(3.2.3.2.3), No. 2(3.2.3.2.5) and No. 1(3.2.3.2.5.1).

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.3.2.6 Control Chain. Where used, control chains shall be of standard aircraft quality and conform to MIL-STD-421. Connecting links shall be retained with standard nonhardened cotter pins. Spring clips shall not be used.

Comparison

The C-5A flight control system does not use any control chain.

Discussion

This is a good requirement which, although not applicable to the C-5A, is applicable to transport type aircraft generally (Lockheed C-130 and C-140). This requirement can be readily demonstrated where applicable and should be specified for future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.3.2.7 Push-pull Flexible Controls. Push-pull flexible controls may be used for transmitting control signals in noncritical applications, but specific approval from the procuring activity must be obtained before use in essential and flight phase essential applications. Where used, they shall conform to MIL-C-7958. Installations shall avoid an excessive number of bends to keep friction forces within acceptable values and minimize the possibility of jamming, and the routing shall preclude damage due to personnel using them as steps and handholds. Conduits shall be supported at frequent intervals, but not so tightly that the control is restrained axially.

Comparison

The C-5A secondary controls used a short length of push-pull flexible control in the fuel shut-off system. This application was believed to be justified because it was a secondary function and the conventional mechanical system would have been more complex and heavier. The system used a "Controlex" ball-bearing type of push-pull flexible cable because of its higher efficiency. The cable run was relatively straight with only two generous radii bends and was supported at frequent intervals. The push-pull flexible control system was designed to the requirements of MIL-C-7958.

Discussion

The push-pull flexible control systems should only be used in very specialized applications and follow the type of guidelines defined in the "Users Guide." This is a good requirement which has been satisfied by the C-5A control system design and compliance can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.3.3 Electrical Signal Transmission. The following requirements apply to all essential and flight phase essential signal paths. Except for power sources, such systems shall be independent of failure modes associated with any other electrical system. Cross connections between redundant electrical signal paths shall be eliminated, or minimized and electrically isolated. Wire runs and components in redundant control paths shall be physically separated and electrical shielding shall be installed, as necessary, to meet failure immunity and invulnerability requirements. All interconnecting wiring shall be prefabricated, jacketed cable assemblies. The outer jackets shall be identifiable by a unique color or other means. Wiring installation shall be in accordance with MIL-W-5088.

Comparison

This requirement is applicable to the C-5A. The C-5A does comply with the stated requirements for electrical signal transmission as the aircraft implements the criteria of MIL-W-5088 and CEI Specification CP 40002. These specifications have more stringent requirements than the above paragraph in some areas.

Discussion

The requirement is qualitative rather than quantitative. This, however, does not distract from the intent of the paragraph which is clearly to insure minimum susceptibility to multiple failures in the transmission signal paths. For this reason the requirement is applicable to transport aircraft and should not be restricted to "essential and flight phase essential signal paths."

Recommendation

It is recommended that the paragraph be changed to eliminate the first sentence and therefore to read as follows in order to increase applicability.

> "3.2.3.3 Electrical Signal Transmission. Except for power sources, such systems shall be independent of failure modes associated with any other electrical system. Cross connections between redundant electrical signal paths shall be eliminated, or minimized and electrically isolated. Wire runs and components in redundant control paths shall be physically separated and electrical shielding shall be installed, as necessary, to meet failure immunity and invulnerability requirements. All interconnecting wiring shall be prefabricated, jacketed cable assemblies. The outer jackets shall be identifiable by a unique color or other means. Wiring installation shall be in accordance with MIL-W-5088."

Requirement

3.2.3.3.1 Electrical Flight Control (EFC) Interconnections. EFC (6.6) wiring in individual channels shall be routed, isolated and protected to minimize the applicable threats to redundancy. Channel loss due to any foreseeable hazard, not extremely remote, shall be limited to a maximum of a single channel. The adequacy of the separation, isolation and protection attainable in any given location for any given hazard shall be evaluated for each aircraft design. Additional protection shall be provided for the EFC wiring where analysis shows that any single hazardous event, not extremely remote, could cause the loss of more than one EFC channel. Primary structural components shall be used to afford this protection where possible. Where it is approved by the procuring activity to route the EFC wiring through wheel wells or other areas subjected, during flight, to the slipstream or impingement of runway fluids, gravel, etc., the wiring shall be protected by enclosures and routed directly through without unnecessary termination or junctions. Where terminations or junctions to equipment in these areas are required, they shall be protected from such impingements. This shall also be done in areas where a high level of maintenance is likely to be required on other systems and equipment.

3.2.3.3.1.1 Cable Assembly Design and Construction. The outer jacketing for EFC wiring shall not create stresses on the wire and connector terminations and shall not stress the wires in a manner which opens the connector grommet seals. During design of the cable assemblies, particular attention shall be paid to the requirements of the circuits within the cable and adequate EMI and EMP control methods, e.g., shielding, twisting, etc., shall be incorporated into the design. Where shielded wires are used, provisions shall be made for carrying the shields through the connectors where single point grounding is necessary. A signal return wire shall be provided for each signal level circuit in the cables. All cable assemblies shall be constructed in an area with temperature and humidity controls and positive pressure ventilation and shall be cleaned (all wire cuttings, etc., removed) and inspected after layup and prior to jacketing to assure that no potentially damaging particles have been included, particularly at the entrance to the grommet seal. All cable assemblies shall be constructed, tested and inspected by specially trained and certified personnel. Terminal boards shall not be used in EFC wiring. Splices shall be qualified, permanent-type splices.

3.2.3.3.1.2 Wire Terminations. Crimp type wire terminations (spade, lug or connector) shall be used on all EFC cables. Soldered and potted connections shall not be used. With the terminal installed on the wire, the wire shall be visible for inspection at both ends of the crimp barrel. The length of wire visible between the insulation and barrel shall not exceed 1/16 inch.

3.2.3.3.1.3 Inspection and Replacement. The EFC wiring shall be installed so that it can be inspected for damage and replaced as necessary. The installation shall provide for visual inspection in critical areas such as hazardous

environment areas or areas where a high level of maintenance is required on system or equipment in close proximity.

Comparison

The C-5A EFC wiring is designed to satisfy the requirements of CEI Specification CP 40002 and military specification MIL-W-5088. The following characteristics are incorporated within the requirements of these specifications.

- Redundant systems must be isolated physically and electrically so that a single failure cannot cause loss of both systems.
- Cables and junctions are identified per standard techniques.
- Wires or cables are not to be routed under or within six inches of flammable fluid lines unless specific precautions are taken.
- Wires and cables are to be routed for maximum reliability and minimum interference and coupling between systems.
- Wires and cables are to be bundled and routed for ease of installation, inspection, and maintenance.
- Crimp type wire connectors are to be used rather than solder.
- Connector grommet seals are to be permanent.
- Only qualified permanent splices are to be used and only where permitted.
- All junctions which are exposed to possible impingement are to be protected with insulating materials.

Discussion

The C-5A complies with the above requirements at least partially. Full compliance is somewhat difficult to ascertain as the requirements are not easily demonstrable. The intention of the requirements is interpreted to be a qualitative statement pertaining to safety of flight through the proper design and installation of the EFC systems. Therefore, the requirement is valid and clearly applicable to transport aircraft. Although there is no reference to MIL-W-5088 in the paragraphs, much of the requirement is contained in that specification which is a general requirement in Paragraph 2.1. No changes are suggested to maintain validity with future transport designs.

Recommendation

Accept "as is."

Requirement

3.2.3.3.2 Multiplexing. Multiplexed signal transmission circuits shall be the digital time-division-multiplexing type utilizing a twisted shielded pair cable as the transmission media for the multiplex bus. The multiplex data bus line and its interface electronics, multiplex terminal unit shall meet MIL-STD-1553.

Comparison

The C-5A aircraft does not have multiplexing in the FCS in either the manual or automatic channels. Some multiplexing is used in other systems on the aircraft.

Discussion

The requirement is not applicable to the C-5A aircraft FCS in particular although it may pertain to transport aircraft in general. Therefore no recommendation is made.

Recommendation

Accept "as is."

Requirement

3.2.4 Signal Computation

3.2.4.1 General Requirements

3.2.4.1.1 Transient Power Effects. Flight control computers shall not suffer adverse effects, which result in operation below FCS Operational State I, due to power source variations within the limits specified for the applicable power system. In the event of power source interruption, no adverse effects shall result which limit operation or performance of flight control computers upon resumption of normal quality power.

Comparison

The C-5A was designed to operate normally with power source variations of MIL-STD-704 power systems. The C-5A Automatic Flight Control Subsystem includes Stability Augmentation Systems which have monitoring for automatic disengagement and positive manual disengagement. Both of these design features can be momentarily affected by power interruptions which, depending upon the length of the interruption, may result in disengagement. Normal operation or performance would not be permanently affected and the pilot can reengage the systems upon resumption of normal quality power.

The C-5A meets the intent of this requirement.

Discussion

If the statement in the requirement, "no adverse effects shall result which limit operation or performance," can be considered to permit system disengagement as normal when due to a power source interruption, then this requirement is valid for present and future aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.4.1.2 Interchangeability. The requirements of 3.2.7.1.2 shall be met, and tolerances shall be such that interchange of any computer component, module, or LRU with any other part bearing the same part number shall require only minimum resetting of parameters or readjustment of other components in order to maintain overall tolerances.

Comparison

The C-5A FCS requirements do not permit resetting of parameters or readjustment of other components on the aircraft in order to maintain overall tolerances. Initial alignment of LRU's is permitted by rigging and null adjustments of electromechanical sensors. Bench adjustment of individual LRU's is permitted in order for the LRU to meet individual acceptance test requirement, but these adjustments must be secured and be tamper-proof prior to installation on the aircraft.

The C-5A meets the intent of this requirement.

Discussion

This requirement is in conflict with requirements 3.1.9.5 and 3.2.7.1.1. Requirement 3.1.9.5 (Invulnerability to Maintenance Error) states that "All line replaceable units (LRU's) shall be designed to permit making internal adjustments only on the bench. The system shall require only a minimum of rerigging following replacement of LRU's." Requirement 3.2.7.1.1 (Standardization) states that "tolerances shall be such that interchange of any LRU with any other part bearing the same part number shall not require resetting of parameters or readjustment of other components in order to maintain overall tolerances and performance." Both these requirements are in line with current electronics technology.

This requirement as written should be deleted and replaced with a new requirement which differentiates between on-aircraft and bench adjustments and allows one-time null adjustments on electromechanical units during installation. The null adjustment could be interpreted as being part of the rigging allowed in Requirement 3.1.9.5. Since null adjustments, rigging and parameter adjustments can be confused, their requirements should be discussed individually. Therefore, changes to the stated requirement are recommended.

Recommendation

Delete the requirement and replace with the following:

"3.2.4.1.2 Calibration Adjustments. The equipment may contain internal adjustments to be used for bench calibration. All equipment shall meet the requirements of 3.1.9.5, 3.2.7.1.1 and 3.2.7.1.2 wherein adjustments of parameters are not permitted on the aircraft. Null adjustment of electromechanical sensors is permitted during rigging,

but should be kept to a minimum. When internal adjustments are used they shall be provided with a positive locking means to prevent any change in adjustment due to environmental conditions encountered in service. Any adjustments which are visible and accessible after assembly shall be tamper-proofed or provided with a seal or other means of determining visually that the adjustments have not been altered."

Requirement

3.2.4.1.3 Computer Signals

3.2.4.1.3.1 Signal Transmissions. Signal transmissions between computer components and modules shall be done by using direct mechanical, hydraulic, pneumatic, or electrical connections, as required. Use of light transmission technology or other nonconventional transmission paths requires specific approval of the procuring activity.

Comparison

Signal transmission between computer components used in, or interfacing with the C-5A FCS, is accomplished by electrical, hydraulic, and/or mechanical means. Further descriptions and comments are contained in the validation discussions under Paragraphs 3.2.4.2 and 3.2.4.3.

Discussion

Throughout the validation of subparagraphs under 3.2.4, as well as other areas, there is a need to clarify or define much of the terminology to avoid confusion and assure a uniformity of understanding. Therefore, definitions of words which are applicable to this and in some cases other areas of validation are recommended for inclusion in Paragraph 6.6. Definitions will be added for the following words: Computer, Mechanical Signal Computation, Module, Component, Circuit, and LRU.

The C-5A FCS complies with the intent of this paragraph and can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.4.1.3.2 Signal Path Protection. Where redundant computing paths are provided they shall be isolated or separated when required to meet the invulnerability requirements of 3.1.9.

Comparison

The C-5A flight controls contain redundant elements in the primary flight controls, some autopilot modes, automatic throttle, angle of attack, stallimiter and stability augmentation systems. These systems were designed to meet the environmental, maintenance, human factors, and failure criteria required for the C-5A. The C-5A requirements are similar to the invulnerability requirements of Paragraph 3.1.9 of this specification. It is felt that the C-5A meets the intent of this requirement.

Discussion

This requirement is valid because it requires that all design parameters be considered in the design of redundant systems. There should be no difficulty in demonstrating compliance with this requirement.

Recommendation

Retain the requirement as stated.

Requirement

3.2.4.2 Mechanical Signal Computation

3.2.4.2.1 Element Loads. Mechanical computer signal transmission elements subjected to the pilots' input force shall be capable of withstanding the loads specified in 3.2.3.2.1.

3.2.4.2.2 Geared Mechanisms. All geared mechanisms used in mechanical computer components shall meet the requirements of MIL-G-6641.

3.2.4.2.3 Hydraulic Elements. Hydraulic computing elements shall be designed in accordance with MIL-C-5503, MIL-H-8775, MIL-C-8890 or ARP 1281 as applicable. MIL-V-27162 shall be used as a general guide for the design of control valves used in hydraulic computing components.

3.2.4.2.4 Pneumatic Elements. All pneumatic computing elements shall be designed in accordance with MIL-P-8564 and AFSC Design Handbook DH 1-6, Section 3G, Pressurization and Pneumatic Systems, as applicable.

Comparison

The term Mechanical Signal Computation is subject to interpretation as to what it encompasses. If we presume that the mechanical computer is a component or sub-system which gathers intelligence and then translates this information into a series of mechanical computing functions to produce another logical output function, then the C-5A elevator artificial feel system could be included. The C-5A elevator artificial feel system shown in Figure 1 (3.2.4.2) uses the variable feel unit shown in Figure 2 (3.2.4.2) as the primary "computing" function. The elevator artificial feel system provides the pilots with artificial feel forces which permit safe maneuvering of the aircraft throughout its operational flight envelope. The total feel system consists of three force producing sources which are springs (centering and servo) input system, bobweight effects, and the VFU system itself. The VFU receives pitot pressure which varies as a function of aircraft speed. This pitot pressure variation moves a hydraulic control valve which varies the magnitude of the hydraulically imposed load acting on a mechanical cam roller. This cam roller rides in a cam attached to a bellcrank which is rotated as a function of the pilots' input motion. Increasing aircraft speed, increases the pitot pressure which increases the hydraulic pressure which simultaneously increases the cam roller force thereby producing a higher "feel" force on the pilots' control column at any position.

The input levers which were subjected to the pilots' input force were designed to withstand the limit loads imposed as specified in Paragraph 3.2.3.2.1. The hydraulic computing elements were designed to the requirements of MIL-C-5503 and MIL-H-8775.

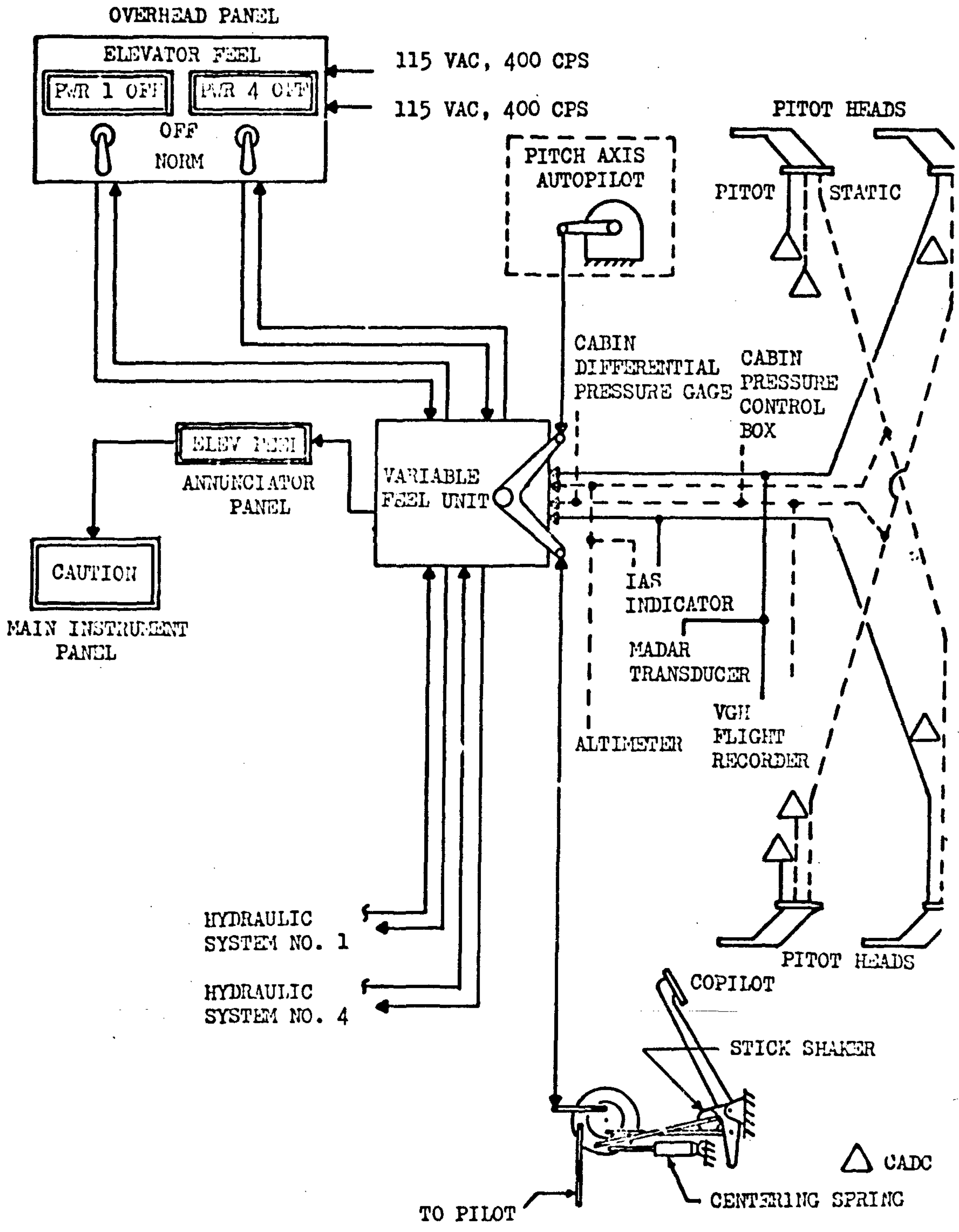
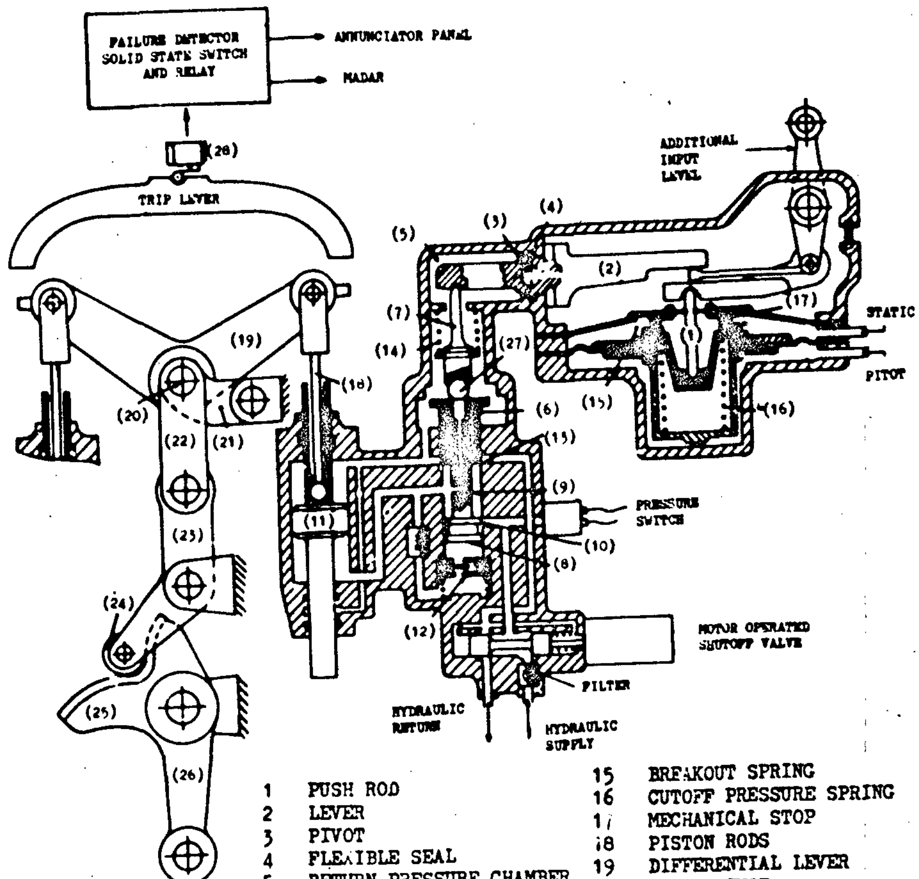


FIGURE 1 (3.2.4.2) ELEVATOR ARTIFICIAL FEEL FUNCTIONAL SCHEMATIC



1	PUSH ROD	15	BREAKOUT SPRING
2	LEVER	16	CUTOFF PRESSURE SPRING
3	PIVOT	17	MECHANICAL STOP
4	FLEXIBLE SEAL	18	PISTON RODS
5	RETURN PRESSURE CHAMBER	19	DIFFERENTIAL LEVER
6	CONTROL VALVE	20	LEVER PIVOT
7	PUSH ROD	21	BELLCRANK LEVER ARM
8	VALVE END FACE	22	CONNECTING LINK
9	SUPPLY PRESSURE PORT	23	CONNECTING LINK
10	SUPPLY PRESSURE PORT	24	ROLLER
11	ACTUATOR PISTON	25	CAM
12	DAMPING ORIFICES	26	OUTPUT SHAFT
13	RETURN PORT	27	RELIEF VALVE
14	BASE PRESSURE SPRING	28	MICROSWITCH

FIGURE 2 (3.2.4.2) VARIABLE FEEL UNIT SCHEMATIC

Another C-5A device using mechanical signal computation is the Central Air Data Computer (CADC) which is a mechanical analog computer. The CADC supplies control signals to the automatic flight control system as well as to other aircraft functional systems. The CADC was designed to the requirements of Contract End Item specification CP 40002-9B which required CADC performance in accordance with MIL-C-38037 except for differences required by the air vehicle requirements. The electronic equipment was designed in accordance with MIL-E-5400, but the mechanical design requirements were not very definitive.

Discussion

Although the C-5A mechanical signal computation elements were not specifically designed to the MIL-F-9490D requirements, these new requirements were met in some instances as discussed above.

Lockheed recognizes a need to clarify what constitutes "mechanical signal computation." It is recommended that more definitive examples be added to the "users guide" along with definitions to distinguish the guidelines of the various computer technologies. These are good requirements which can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is" and expand the "users guide" to give more definitive examples of "mechanical signal computation."

Requirement

3.2.4.3 Electrical Signal Computation

3.2.4.3.1 Analog Computation. Redundant electrical signal paths within a computer shall be isolated as required by failure immunity and invulnerability requirements specified herein. For failures which may cause a hazardous deviation in the aircraft flight path, the computer shall have provisions for rapidly disabling its command outputs or servos unless other fail-safe provisions exist.

Comparison

The C-5A flight control computers which contain redundant signal paths contain monitoring to insure proper operation. For those systems which control the aircraft motion, automatic disengagement will occur for failures within the computer which can cause changes in the flight path. At no time can a failure occur that will cause a hazardous deviation of the aircraft flight path.

The C-5A flight control system meets the intent of this requirement.

Discussion

This is a good requirement and is relevant for future aircraft. This same type requirement should be imposed on digital computers.

Recommendation

Retain the requirement as stated.

Requirement

3.2.4.3.2 Digital Computation. At the time of aircraft acceptance by the procuring activity, the total time used in flight control computations for worst case conditions shall not exceed 75 percent of the available computation time allocated for flight control use. Resident and bulk storage shall be sized such that at least 25 percent of each type is available for growth at the time of aircraft acceptance. Computation and sample rate shall be established at a level which ensures that the digital computation process will not introduce unacceptable phase shift, round off error, nonlinear characteristics, and frequency foldover or aliasing into the system response.

Comparison

This requirement was not applicable to the C-5A FCS design since digital computation was not used.

Discussion

This requirement is expressed well and concisely. Quite sensibly, it avoids undue restrictions which might hamper innovative digital control law implementation. The reserve computation time and memory requirements are vital for practical systems, not only for growth, but to alleviate undue emphasis upon programming efficiency caused by marginal capacity. The sampling rate requirement in essence states that the sampling process should not cause unacceptable effects, and no merit is seen is trying to be more specific.

Recommendation

Revise the requirement as follows. Add the following sentence:

> "The failure immunity and safety requirements of Paragraph 3.1.3.2 shall be met."

Requirement

3.2.4.3.2.1 Memory Protection. Memory protection features shall be provided to avoid inadvertent alternation of memory contents. Memory protection shall be such that neither electrical power source transients within the limits specified nor EMI as specified in 3.2.5.4.1 shall cause loss of program memory, memory scramble, erroneous commands, or loss of ability for continued operation. The transients shall be as specified in MIL-STD-704 for Category C utilization equipment. For applications where system failures could be hazardous to safety of flight, the levels for normal, abnormal, and emergency electric system operation shall apply. For applications which are not critical to safety of flight, the levels for normal operation shall apply. These transient requirements shall apply to cases when all or only one of the redundant power sources are operating.

Comparison

Not applicable to C-5A.

Discussion

This requirement is satisfactory and requires no editing. It is specific enough to foster good design and general enough not to restrict design options.

Recommendation

Retain the requirement as stated.

Requirement

3.2.4.3.2.2 Program Scaling. Parameter scaling, word size, input limiting, and overflow protection shall ensure correct processing and continuous safe operation for all possible combinations of maneuvering demand and gust or other plausible disturbance within the service envelope of the system. Any condition capable of producing an overflow in an essential or flight phase essential function shall be precluded by hardware overflow detection and software or firmware that provides for data recovery and continuous safe operation following an overflow. Scaling shall provide satisfactory resolution to prevent the granularity due to digitizing processes from introducing, into the system response, unacceptable levels of nonlinear characteristics or instabilities.

Comparison

Not applicable to the C-5A.

Discussion

This requirement is satisfactory and requires no editing. It is specific enough to foster good design and general enough not to restrict design options.

Recommendation

Retain the requirement as stated.

Requirement

3.2.4.3.2.3 Software Support. For programmable computers a software support package shall be provided to aid in generation and validation of new programs. This support package shall be designed to be executable, either on the airborne computing system for which it was designed or on a large scale digital computer specified by the procuring agency. The support package shall include the necessary software and appropriate peripheral devices in accordance with the contractor data requirements list (DD 1423).

Comparison

Not applicable to the C-5A.

Discussion

The specification of a large scale digital computer for executing the support software package seems to be an unwarranted restriction. It may in many cases be more economical and feasible to use a smaller computer.

Recommendation

Delete "large scale" from the second sentence.

Requirement

3.2.5 Control Power

3.2.5.1 Power Capacity. Sufficient electrical, hydraulic, and pneumatic power capacity shall be provided in all flight phases and with all corresponding engine speed settings such that the probability of losing the capability to maintain at least FCS Operational State III airplane performance shall be not greater than extremely remote when considering the combined probability of system and component failure and the cumulative exceedance probability of turbulence. Hydraulic power shall be used to actuate powered essential and flight phase essential MFCS.

Comparison

The C-5A FCS was designed to have sufficient power capability to meet the redundancy requirements of CEI CP 40002-6B and MA0001A air vehicle flight controls subsystem requirements. As noted in the validation discussion for Paragraph 3.1.3.1 "Redundancy" the FCS was designed to permit continued operation after any single malfunction, to provide aircraft controllability after the loss of two hydraulic systems and to provide a redundant design wherever a failure could involve safety of flight.

The level of redundancy is indicated by the validation discussion for Paragraph 1.2.3 FCS Criticality Classification and in Table 4 which shows the effects of multiple hydraulic system failures on flight controls. It can be concluded that no single failure in a subsystem providing a control function will cause degradation of the aircraft below Operational State II. The hydraulic system distribution for the FCS is shown in Figure No. 1 (3.1.8.1) and discussed in the validation for Paragraph 3.2.5.3. Similar redundancy levels are provided so that multiple electrical power losses are required as shown in Table 5 to cause degradation of the aircraft below Operational State II.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

TABLE 4. HYDRAULIC SYSTEM FAILURE EFFECTS ON FCS

HYDRAULIC SYSTEM MULTIPLE FAILURE CONDITIONS:

1. Combined Loss of Systems 1 and 2
2. Combined Loss of Systems 2 and 3
3. Combined Loss of Systems 1, 2 and 3
4. Combined Loss of Systems 1, 3 and 4
5. Combined Loss of Systems 2, 3 and 4

	MEETS OPERATIONAL STATE				
	I	II	III	IV	V
o LIFT					
T.E. Flaps			1,3,4		
L.E. Slats			1,3,4		
Ground Spoiler	2				
o PITCH					
Outboard Elev.		1,2		4	3
Inboard Elev.		1,2		4	3
Stab. Aug.	1				
ALDCS	1,2				
Trim		1,2,4			3
o ROLL					
Ailerons		2	1	3,4	
Spoilers/Roll		2	1	3,4	
Stab. Aug.		1,2			
ALDCS	1,2				
Trim					
o YAW					
Upper Rudder			1,2	4,3	
Lower Rudder			1,2	4,3	
Stab. Aug.		1,2		3	
Trim					
Limiter					

TABLE 5. ELECTRICAL SYSTEM FAILURE EFFECTS ON FCS

ELECTRICAL FAILURE CONDITIONS

1. One System Failure/Loss
2. Two System Failure/Loss
3. Three System Failure/Loss

	MEETS OPERATIONAL STATE				
	I	II	III	IV	V
o LIFT					
T.E. Flaps		1	2		
L.E. Slats		1	2		
Ground Spoiler		1	2,3		
o PITCH					
Outboard Elev.		1,2	3		
Inboard Elev.	1		2		
Stab. Aug.		1,2			
ALDCS	1,2	1	2		
o ROLL					
Ailerons		1,2	3		
Spoilers/Roll		1,2	3		
Stab. Aug.		1,2	3		
ALDCS	1,2				
Trim	1,2				
o YAW					
Upper Rudder		1	2		
Lower Rudder		1	2		
Stab. Aug.		1,2			
Trim	1,2				
Limiter	1,2				
o AUTO THROTTLE	1				
o GO-AROUND	1				
o FLIGHT DIRECTOR	1				
o STALLIMITER		2			
o AUTOPILOT		1			

Requirement

3.2.5.2 Priority. Essential and flight phase essential flight controls shall be given priority over noncritical controls and other actuated functions during simultaneous demand operation. However, no specific priority provisions, such as hydraulic priority valves, are required unless there is a likelihood of simultaneous demands which could prevent one or more essential or flight phase essential actuation systems from meeting their performance requirements. Where provided, priority controls shall be highly resistant to deterioration, binding, or failure while dormant under normal aircraft operations so that they will function as required when conditions dictate. If flight safety can be endangered by failure of such controls, ground checkout means for ready determination of their operability shall be provided and procedures specified.

Comparison

The primary actuation power for the C-5A FCS is hydraulic. Hydraulic power distribution to the FCS is shown in Figure No. II-2. The FCS is provided with enough hydraulic flow capability to supply any probable simultaneous hydraulic flow demand with enough margin to maintain control of the aircraft. Figure No. 1 (3.2.5.2) shows the maximum hydraulic flow demands and capability for the FCS and all utility operations for the aircraft operation. It is highly improbable that all the flight control functions will be operated simultaneously at maximum rate. Normally the flaps, pitch trim and landing gear are sequenced and any other functions using maximum rate would at the worst be exposed to small and brief transient pressure drops. In the event of loss of normal hydraulic power supplies to a critical FCS function, emergency hydraulic power is supplied by the hydraulic power transfer units. These provide power for one hydraulic system from another without interchange of hydraulic fluid. Hydraulic shut-off and power transfer provisions are subjected to periodic inspections and functional testing on the aircraft to assure their availability when needed.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

SYSTEM		MAX. FLOW DEMANDS -G.P.M. SYSTEM 1	SYSTEM 2	SYSTEM 3	SYSTEM 4	TOTAL
PRIMARY FLIGHT CONTROLS						
ELEVATOR	INBD	--	20.4	20.4	--	40.8
	OUTBD	4.1	4.1	4.1	--	12.3
RUDDER	LOWER	--	6.5	6.5	--	13.0
	UPPER	4.9	--	4.9	--	9.8
FLIGHT SPOILERS	FLAPS UP	16.2	24.3	24.3	16.2	81.0
	FLAPS DOWN	19.5	27.2	27.2	19.5	93.4
AILERON		13.0	26.0	--	13.0	52.0
PITCH TRIM		6.0	26.0	--	--	[illegible]
SECONDARY FLIGHT CONTROLS						
TRAIL EDGE FLAPS AND LEAD EDGE SLATS			--	--		
PITCH "Q" FEEL		0.5	--	--	0.5	1.0
			--	--		
STALLIMETER		0.5	--	--	0.5	1.0
UTILITY & OTHER OPERATIONS						
THRUST REVERSERS		15.5	15.5	15.5	15.5	62.0
GROUND SPOILERS	PANELS 1,2,3,4	29.4	--	--	37.6	67.0
	PANELS 5,6,7,8,9	32.4	58.6	58.6	32.4	182.0
LANDING GEAR		60.0	--	--	40.0	100.0
FWD LOADING		--	--	--	13.5	13.5
AFT LOADING (ADS)		30.5	--	--	--	30.5
AIR COMPRESSOR		--	--	--	1.7	1.7
AUX GENERATOR			10.2	--	--	10.2
AERIAL REFUELING		--	--	--	4.0	4.0

C-5A FLOW (GPM) CAPABILITIES

SOURCE		SYS 1	SYS 2	SYS 3	SYS 4
ENGINE DRIVEN PUMPS (2 PER SYSTEM)	IDLE	80	80	80	80
	CRUISE	120	120	120	120
	TAKE OFF	128	128	123	128
ATM PUMP		40	---	---	40
PWR TRANSFER UNIT		35	35	35	35

FIGURE 1 (3.2.5.2). HYDRAULIC SYSTEM FLOW DEMANDS AND CAPABILITIES

Requirement

3.2.5.3 Hydraulic Power Subsystems. All hydraulic power generated and distribution systems normally used for flight control shall be designed in accordance with MIL-H-5440 and MIL-H-8891 as applicable. The FCS shall operate in accordance with this specification when supplied with such hydraulic power. Applicable requirements in AFSC Design Handbook DH 1-6, Systems Safety, Section 3F, Hydraulic Systems, shall also be met.

Comparison

The C-5A hydraulic power subsystems used to supply the flight control system components were designed in accordance with the requirements of Contract End Item (CEI) specification CP 40002-5B and conform to the requirements of MIL-H-5440 to the extent specified in the CEI document.

Four independent, engine-driven, 3000 psi, closed center hydraulic systems distribute power to the flight control components as shown in Figure 1 (3 1.8.1) and Figure II-2. Two independently powered systems are used exclusively for the pitch, roll, and yaw flight control systems. The other two systems are used for those same controls; for flaps, slats and ground spoiler controls; and other hydraulic power functions. Each flight control surface is powered by two hydraulic systems which are each capable of supplying full system rate and one-half hinge moment, except for the outboard elevator which is powered by three systems providing one-third hinge moment each. The hydraulic power transfer units use power from one system to provide power in another system in the event of power loss. The redundancy requirements for all the FCS systems have been satisfied as noted in Paragraph 3.1.3.1 by virtue of the number and distribution of the hydraulic systems and their power supplies.

In the event of loss of hydraulic power on all engines, a ram air turbine supplies enough hydraulic power to the critical FCS for Operational State IV. Recommended requirements for tubing, mounting, seals, components, filtering, pressure and operational tests, etc., have been met. Hydraulic system temperature and pressure controls have been provided as required. Failure isolation was provided by devices such as shut-off valves, check valves, reservoir level sensing.

Discussion

This is a good requirement which has been satisfied by the C-5A, can be readily demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.5.4 Electrical power subsystems. All electrical power generation and distribution subsystems used for flight control shall provide electrical power in accordance with MIL-STD-704. The FCS shall operate in accordance with this specification when supplied with power in accordance with MIL-STD-704. Applicable requirements in the following AFSC design handbooks shall be met:

a. DH 1-4: Electromagnetic Compatibility.

b. DH 1-6: System Safety.

c. DH 2-1: Airframe

d. DH 2-2: Crew Stations and Passenger Accommodations.

Electrical systems which provide power to essential or flight phase essential controls, shall insure uninterruptible, isolated redundant power of adequate quality to meet FCS requirements after any malfunction not considered extremely remote. Such electrical systems shall, except for basic power source, be independent of failure modes associated with any other electrical system. Essential and flight phase essential FCS shall be automatically provided alternate sources of power where interruption could result in operation below FCS Operational State III. A protected alternate source of power shall be provided for all essential or flight phase essential control signal transmission paths sufficient to continuously maintain at least FCS Operational State III performance in the event of loss of all electrical power supplied from engine-driven generators. Control systems employing both ac and dc power inputs shall normally have interlocks incorporated to disconnect both power inputs should either type of power be lost. However, if the loss of either power source can be shown to be equivalent to loss of both or FCS Operational State III or better is maintained with either power source, interlocks are not required.

Comparison

The electrical power generation and distribution subsystems used to supply the C-5A FCS were designed to the requirements of CEI specifications CP 40002-1A, -5B, -6B, and MIL-E-25499, to the extent specified therein. Electrical power for the FCS includes 115/200 VAC, 400 cycle (HZ), 3-phase and 28 VDC in accordance with the limits of MIL-STD-704. Electrical power is used in the FCS for functions such as:

- Operation of hydraulic servo shut-off valves
- Operation of electro-mechanical actuators used for autopilot, autothrottle, trim, feel, and FCS limiting
- Operation of dedicated displays

The primary electrical power source is provided by engine-driven, constant-speed drive, brushless generators operating in parallel. A.C. power is individually regulated to the required voltages. Generator and bus tie contactors were in accordance with MIL-R-6106. A.C. power distribution is provided by an individual load bus from each generator contactor. D.C. power is derived from A.C. powered transformer-rectifier units supplied thru two main D.C. load buses.

In-flight emergency power is supplied thru electrically and physically independent emergency A.C. and D.C. load buses.

A gas turbine powered auxiliary power unit (APU) supplied auxiliary power. The APU is normally used on the ground prior to engine start up; however, it has in-flight start and operation capability.

In the event of a four-engine-out incident, limited in-flight electrical power for emergency control is provided by deployment of a Ram Air Turbine (RAT) generating unit.

Electromagnetic compatibility limits were in accordance with MIL-E-6051 and MIL-Std-826 as noted in the validation discussion for Paragraph 3.2.5.4.1.

The electrical distribution system insures uninterrupted redundant power to meet the FCS operational requirements commensurate with the required failure criteria. The power distribution system has A.C. and D.C. interlocks and emergency power buses incorporated which assure proper emergency power distribution to the critical control functions in the event of power loss on the primary distribution system. In any event the loss of electrical power will not degrade the aircraft controllability below operational State III.

Discussion

This is a valid requirement which has essentially been met by the C-5A design. The requirement can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.5.4.1 Electromagnetic Interference Limits. The FCS shall operate within the limits of MIL-E-6051 and MIL-STD-461 environment. Electromagnetic interference created by the systems and components during normal operation shall be within the limits of MIL-E-6051 and MIL-STD-461, respectively. Failure modes of all onboard systems and equipment, including flight controls, wherein these limits may be exceeded shall be identified in addition to sources of conducted EMI that may be detrimental to FCS operation. Additionally, the estimated magnitude of EMI generated by these failure modes shall be provided for the assessment of the safety of the EFCS.

Comparison

The C-5A FCS was designed to meet MIL-E-6051 and MIL-STD-826 requirements. MIL-STD-461 supersedes MIL-STD-826. During development and testing of the AFCS changes were required in meeting MIL-STD-826. Some of the areas were redesigned, but where it was impractical to redesign and where flight safety or normal operation was not affected, deviations were obtained from the customer. MIL-STD-461 does impose different requirements than MIL-STD-826. It is felt that any problems with meeting this new specification would have to be broached with each particular design. The last two sentences were not required for the C-5A design and therefore are not met.

Discussion

The first two sentences of this requirement where MIL-E-6051 and MIL-STD-461 are imposed are considered reasonable, but the last two sentences create a very impracticable requirement. It would be almost impossible to failure analyze every component on the aircraft to determine if its failure would create a detrimental condition to the FCS in the area of EMI or by the FCS to other systems. In addition, to estimate the magnitude of the EMI generated by these failures, if they could be identified, would be a difficult and costly task.

Normally the equipment is designed to the best ability of the designer to meet the requirements. After the design is completed and the unit built, then the equipment is tested to determine its capability of meeting the requirements. Changes are made to the system to correct any major problems which affect aircraft safety, operation or mission accomplishment. This procedure should be adequate for all designs and to impose the additional requirements of failure analysis of all equipment to determine their EMI effect would impart an unnecessary hardship and increase the cost of any new program. The requirement is valid, has been satisfied and can be demonstrated to the extent noted and should be specified for all future transport type aircraft with the recommended change.

Recommendation

Revise the requirement as follows: Delete the last two (2) sentences.

Requirement

3.2.5.4.2 Overload Protection. Overload protection of the primary power wiring to the system or component shall be provided by the airplane contractor. Installation requirements of the system or component specification shall specify the values of starting current versus time, surge currents if applicable, normal operating current, and recommended protective provisions. Additional protection as necessary shall be provided within the system or component. Such circuit protection shall not be provided in signal circuits or other circuits where opening of the protective devices will result in unsafe motion of the aircraft.

Comparison

This requirement is the same as the requirement that the C-5A was designed to meet. Circuit breakers are installed on all power lines going to the FCS. The C-5A meets this requirement.

Discussion

This is a valid requirement which has been satisfied by the C-5A design. It can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.5.4.3 Phase Separation and Polarity Reversal Protection. In systems affecting flight safety, phase reversal and polarity reversal shall be prevented as far as practical by keying, physical restraints or other positive means.

Comparison

The C-5A has been designed to meet this requirement. All connectors are either keyed differently or by use of wire routing and harness ties cannot be normally connected incorrectly.

Discussion

This is a good requirement, but it does not cover the case where phase separation and polarity reversal cannot be protected against the human element. When designing systems that affect flight safety, all elements must be considered, even human errors. An example is the C-5A Horizontal Situation Indicator (HSI) and the Airspeed Vertical Scale Flight Instrument (VSFI). The C-5A VSFI's are basically the units used on the C-141 aircraft but the face was modified for the C-5A missions. Keying and connector size on these units were correlated with all other equipment that were available. When the HSI, which is GFE, was defined it was found that it had the same connector and keying as the airspeed VSFI. By using different routing and length of cable (physical restraint), it made the correct connector only accessible to its instrument.

Even with all these precautions and personnel training an instance of cross connecting the HSI and VSFI connectors occurred during C-5A production. Ties, which were designed to prevent cross connection, were cut to accomplish this cross connection. This condition was discovered during ground checkout. This example indicates the importance of considering the human element in the design of systems especially those that may affect flight safety. It is recommended that a sentence be added to the requirement to insure that any error in connecting must be detectable prior to flight.

Recommendation

Revise the requirement as follows:

> Add the following sentence: "Where it is impossible to assure prevention of incorrect electrical connections, they shall be readily apparent through normal maintenance and inspection procedures, ground checkout procedures (including preflight) or indications in the flight station."

Requirement

3.2.5.5 Pneumatic Power Subsystems. Pneumatic power using ram-air, engine bleed air, stored gas, mechanically compressed air, or generated gas may be used for noncritical flight control functions and for driving hydraulic pumps and electric generators. High pressure pneumatic systems used for FCS functions shall conform to MIL-P-5518, the applicable requirements in AFSC Design Handbook DH 1-6; System Safety, Section 3G; Pressurization and Pneumatic Systems, and the applicable requirements under 3.2.5.1, herein. Engine bleed air systems used for FCS functions shall conform to MIL-E-38453.

Comparison

The C-5A FCS does not use pneumatic power for any of its control actuation functions. The C-5A has a pneumatic system which conforms to MIL-P-5518 that is used as an airborne compressor to accomplish servicing functions such as landing gear struts and hydraulic accumulators. Another pneumatic system is a turbo compressor for inflating landing gear tires.

Discussion

This is a valid requirement which is not applicable to the C-5A FCS; however, it could be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.6 Actuation

3.2.6.1 Load Capability

3.2.6.1.1 Load Capability of Elements Subjected to Pilot Loads. Elements of actuation systems subjected to loads generated by the pilot(s) shall be capable of withstanding the loads due to the pilot's input limits specified in MIL-A-8865, Section 3.7, Flight Control System Loads, taken as limit loads unless higher loads can be imposed such as by a powered actuation system or loads resulting from aerodynamic forces. Control signal boost actuator outputs may be load limited by spring cartridges.

Comparison

The C-5A flight control systems (FCS) and components were designed to meet the strength requirements of Contract End Item (CEI) specification CP 40002-2, Paragraph 3.4.10.5. The CEI design criteria is essentially the same as the requirements specified in MIL-A-8865, Section 3.7. The dual C-5A FCS components and mechanical system transmitting elements were designed to withstand the maximum limit load resulting from a pilot input load equivalent to 75 percent of the dual pilot effort specified in MIL-A-8865, Section 3.7 when applied at the pilots controller and reacted throughout the control system. This design load requirement was reacted via the mechanical signal transmission system as required by 3.2.3.2.1 thru the servo actuator input valve arm into the mechanical stops for the valve arm. All the primary hydraulic servo actuators except the outboard elevator servo have the feedback crank and feedback rod designed to the effects of this pilot limit load. In the outboard elevator servo enough valve overtravel was provided to prevent the feedback summing crank from bottoming out during the extremes of pilot input to the valve in one direction and actuator/surface movement in the opposite direction. Figure 1 (3.2.6.1.1) illustrates a typical primary (aileron) servo actuator input and feedback crank arrangement. Figure 1 (3.1.4) illustrates the typical primary servo input and feedback summing crank within the servo manifold. The input bungees shown in Figures No. 1 (3.2.6.1.1), No. II-4, II-5 and II-7 are there primarily to prevent restriction of the pilot inputs to the other servos when that servo has been deenergized and does not move; however, this was considered to be solid link for the structural design load limits that can be imposed on the servo input/feedback system.

Discussion

The concern with this requirement is the question of what constitutes an "element of the actuation system" as opposed to what the limit is for a "mechanical signal transmission system" as required by 3.2.3.2.1. In the case of the C-5A full power primary servo actuators, the input/feedback servo linkage discussed above is primarily a "valve summing" function for

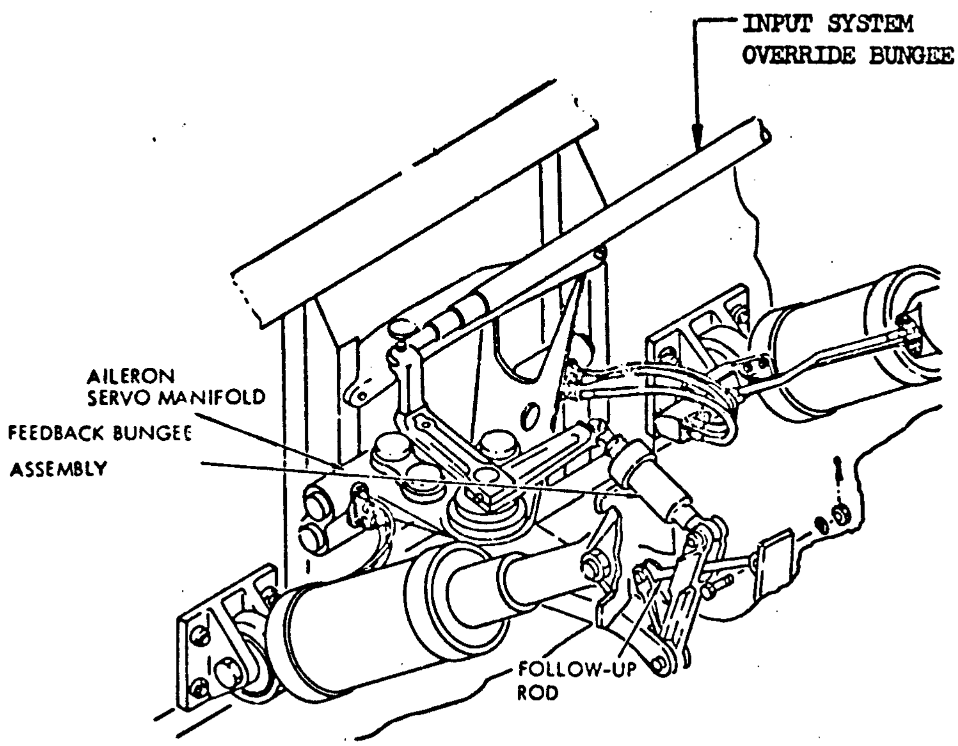


FIGURE NO. 1(3.2.6.1.1) AILERON SERVO ACTUATOR INSTALLATION

the actuation system, but it does not contribute to the actual output hinge moment as part of its normal function. On the other hand, a power boost or mechanical reversion system would have pilot loads reacted to the surface actuation system and it is obvious that the mechanism would have to be designed to the maximum input limit load from either the powered actuation system or from the aerodynamic loads. However, it should be noted that even in the fully powered servo actuation systems the input/feedback mechanisms may be subjected to limit load conditions from pilot input such as when the servo actuator has been de-energized, is not moving, and the linkage bottoms out. It would be beneficial to add more definitive examples of full power servo actuation systems, booster systems, manual reversion systems, etc., in the "Users' Guide" in order to clarify the design guidelines for the use of the word "elements" as it presently is used in 3.2.3.2.1 and 3.2.6.1.1. The last sentence of the requirement is not clear. This is an otherwise good requirement, which can be helped by clarification, and which has been satisfied by the C-5A control system design. It can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Revise the last sentence of the requirement to read as follows:

"Control signals to servo actuators may be load limited."

Additional Data

Pilot induced loads actually encountered by FCS elements may vary greatly depending upon the kinematics of the system, the operating condition (normal/abnormal) of the system, the general type of system, ground gust provisions and airplane maneuvering levels. Static design load requirements for the total input system are, however, essentially the same for all systems, regardless of type when the requirements to relieve a jam are applied. The power boosted and fully powered FCS usually incorporate either surface damping or gust locks into the FCS servoactuators for protection of the airplane structure and FCS against the effects of ground gusts. Mechanical elements of those input systems are therefore not subjected to loading from ground gusts. In pure mechanical or aerodynamically boosted systems either surface dampers or mechanical gust locks may be used. When the ground gust protection is incorporated into the mechanical input system, the elements of that system which are between the protecting device (gust lock or other) and the surface are subject to loading from ground gusts.

In the fully powered FCS using mechanical input provisions for the pilot the C-5A uses override bungees in the input linkage to the servo actuators. These bungees and servo valve overtravel permit operation of normal servoactuators after malfunctioning servoactuators have been shut down. Figures [II-4] and [1(3.2.6.1.1)] for the C-5A roll control system depict appropriate uses for override bungees.

Note: For Figure [No.] use the next available "Users' Guide" number.

Requirement

3.2.6.1.2 Load Capability of Elements Driven by Power Actuators. Elements subjected to loads generated by a powered actuation system, including all parts of the actuator shall be capable of withstanding the maximum output of the actuation system, including loads due to bottoming, or the maximum blowback load, as controlled by pressure relief valves or other load limiting provisions, whichever is greater, as the limit load. Ultimate load capability shall be 1.5 times limit load. In dual load path design, each path shall be capable of sustaining load as specified in 3.1.11.1.2 without failure.

Comparison

The C-5A rudder, roll control, elevator control and ground spoiler systems utilize fully powered hydraulic servos to drive the control surfaces. As shown in Figure II-7, each of the two rudders (upper and lower) is controlled by a servo having two hydraulic actuators operating in parallel and in a force sharing manner. As shown in Figure II-4, the left and right wing ailerons are controlled by a servo having two (2) hydraulic power actuators operating in parallel and in a force sharing manner. Each of the five flight spoilers on each wing is controlled by a tandem servo actuator. The ground spoiler system consists of four ground spoilers and five combination ground and flight spoilers. Extension and retraction of each panel, one (1) through four (4), is accomplished by means of a single tandem actuator that operates open loop (uses no feedback). As shown in Figure II-5, each of the inboard elevator surfaces are controlled by a single power servo having two power actuators operating in parallel and in a force sharing manner. Each of the two outboard elevator surfaces are controlled by two separate servos that receive a common input. One servo has two power actuators and the other servo has a single power actuators and the other servo has a single power actuator, all of which operate in parallel and in a force sharing manner.

Most actuators in these systems contain snubbers at each end of their stroke. Consequently, bottoming loads are nominal and are not critical design factors. The maximum output of each actuator is limited by relief valves. The actuators are designed for limit loads based on a maximum system pressure of 3390 psig and a system return pressure of 175 psig. The structure is designed to withstand ultimate loads of 1.5 times the limit load. In case of failure of one of the dual load paths of the aileron, elevator or rudder servos, the remaining load path can sustain the loads defined in paragraph 3.1.11.1.2.

The C-5A complies with this requirement.

Discussion

The requirement covers the subject well and is applicable to future transport aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.6.2 Mechanical Force Transmitting Actuation. For control cable actuation, the requirements specified in 3.2.3.2.4 and subparagraphs apply. For push-pull rod actuation, the requirements specified in 3.2.3.2.5 and subparagraphs apply.

Comparison

This paragraph addresses requirements already well defined and does not address its subparagraphs. The referenced paragraphs, 3.2.3.2.4 and 3.2.3.2.5 are validated separately.

Discussion

This paragraph, as a lead in to its subparagraphs, should address the requirements included in the subparagraphs.

Recommendation

Completely reword the paragraph as given below:

> 3.2.6.2 Mechanical Force Transmitting Actuation. In addition to the requirements specified in paragraphs 3.2.3.2.4 and 3.2.3.2.5 and their subparagraphs, the following additional requirements apply to mechanical force transmitting actuation systems.

Requirement

3.2.6.2.1 Force Transmitting Powerscrews. Powerscrews with rotary input and linear output motion may be used to actuate relatively low-duty-cycle flight control surfaces, such as wing flaps and trimmable stabilizers. but specific approval from the procuring activity shall be obtained before use in high-duty-cycle applications. Nonjamming mechanical stops shall be provided at both ends of the screw to limit travel of the nut; and, they shall be designed to withstand all possible loads, including possible impact loading, without failure. Provisions shall be incorporated into the nut to minimize entry of sand, dust and other contaminants; to retain its lubricant; and to preclude the entry or retention of water. However, positive sealing is not required if the screw is installed such that it is protected from such contamination or is inherently resistant to wear and jamming by contamination.

Comparison

Powerscrews with rotary input and linear output motion are utilized in the C-5A secondary flight control systems only. These systems are the leading edge slat drive system, trailing edge flap drive system and horizontal stabilizer trim control system.

The leading edge slat system utilizes turnbuckle type ballscrew actuators to extend and retract the slats. The trailing edge flap system utilizes simple fixed length ballscrew actuators to extend and retract the flaps. Each slat actuator utilizes two recirculating ball nuts whereas each flap actuator utilizes a single traversing recirculating ball nut. The normal operating stroke of the actuators in both the slat and flap systems is less than the stop-to-stop stroke capability. Consequently, the mechanical stops, which are the non-jamming type, are not contacted unless a failure has occurred in the position control system. The stops were designed to withstand all system impact loads for a minimum of 48 times at half speed and 2 times at full actuation speed. The stops were tested for compliance to this requirement during the qualification test program. The ball nuts utilize seals and wipers to minimize entry of sand, dust and other contaminants; to retain its lubricant; and preclude the entry or retention of water. Grease fittings are used to lubricate the ball nuts.

The horizontal stabilizer trim control system utilizes a pitch trim actuator which provides a dual structural load path between the horizontal stabilizer and the vertical stabilizer. The mounting for the upper end of the actuator is supported on the front spar of the horizontal stabilizer. The mounting for the lower end of the actuator is supported on the front spar of the vertical stabilizer. The pitch trim actuator utilizes an irreversible acme screw and nut designed on a dual load path basis. A non-jamming mechanical stop is provided at the top and the bottom of the screw to eliminate overtravel of the screw in either direction in the event a failure occurs that would permit actuator travel in excess of the normal stroke. The screw is completely covered with a bellows type boot to prevent the entrance of dust and other contaminants.

An automatic greaser is utilized to periodically pump a small amount of grease into the nut. Due to the use of the boot to protect the screw and the particular nut design, positive sealing is not utilized.

In summary, the force transmitting powerscrews used on the C-5A are in compliance with this requirement.

Discussion

The requirement covers the subject quite well except that it does not include a requirement for the number of impacts the non-jamming stops must be designed for. The requirement would vary with the system design and whether or not the stops would be contacted during normal operation.

Recommendation

Delete the second sentence and replace with the following sentence:

> "Non-jamming mechanical stops shall be provided at both ends of the screw to limit travel of the nut; and, they shall be designed to withstand all possible loads, including those produced by impacting the stops at maximum operational rate, without deformation or failure."

Insert the following sentence between the second and third sentences:

> "Systems incorporating non-jamming stops in actuation components or assemblies shall be designed to preclude their contact during normal operation."

Requirement

3.2.6.2.1.1 Threaded Powerscrews. Standard thread forms only shall be used, and the thread roots shall be rounded as necessary to preclude stress cracking. Lubrication provisions shall be adequate for controlling efficiency, wear, and heating to acceptable values. Where in service lubrication is necessary, lube fittings in accordance with 3.2.7.2.5 shall be provided. If the design is dependent on inherent friction to maintain irreversibility, this characteristic must be adequate under all expected operating conditions, including the full range of loads, both steady loads and reversing or variable-magnitude loads which may be encountered due to control surface buffeting or buzz, temperatures, and environmental vibration over the full service life of the unit.

Comparison

There is only one threaded powerscrew used in the C-5A flight control systems. It is the screw used in the pitch trim actuator assembly shown in Figure 1 (3.2.6.2.1.1). A standard acme thread form is used because of its inherent irreversibility characteristic. The thread roots are sufficiently rounded to preclude stress cracking. As a means of controlling efficiency, wear and heating to acceptable values, an automatic lubrication system is utilized. In-service lubrication of the screw and nut involves only the periodic replacement of the cannister containing the lubricant. Irreversibility of the screw and nut interface was demonstrated under simulated loaded and unloaded operating conditions during the qualification test program. However, as a backup to the inherent irreversibility, the screw and nut drive mechanisms incorporate no-backs that prevent screw or nut rotation during the absence of an operational signal.

In summary, the pitch trim actuator meets the intent of this requirement.

Discussion

This requirement adequately covers the design points that must be addressed relative to the use of power screws in an aircraft flight control system. It is a reasonable requirement which can be demonstrated practically.

Recommendation

It is recommended that this requirement be accepted "as is."

1. Pitch Trim Actuator Assembly
2. Manual Torque Drive
3. Hoisting Eye
4. Bolt Assemblies
5. Hydraulic Connection
6. Input Bellcrank
7. Bellcrank Support
8. Nut Drive Input Rod
9. Electrical Connector
10. Grease Tube
11. Coupling Nut
12. Automatic Greaser
13. Nut Drive Motor
14. Nut Drive Manifold
15. Screw Drive Motor
16. Screw Drive Manifold
17. Control Cables

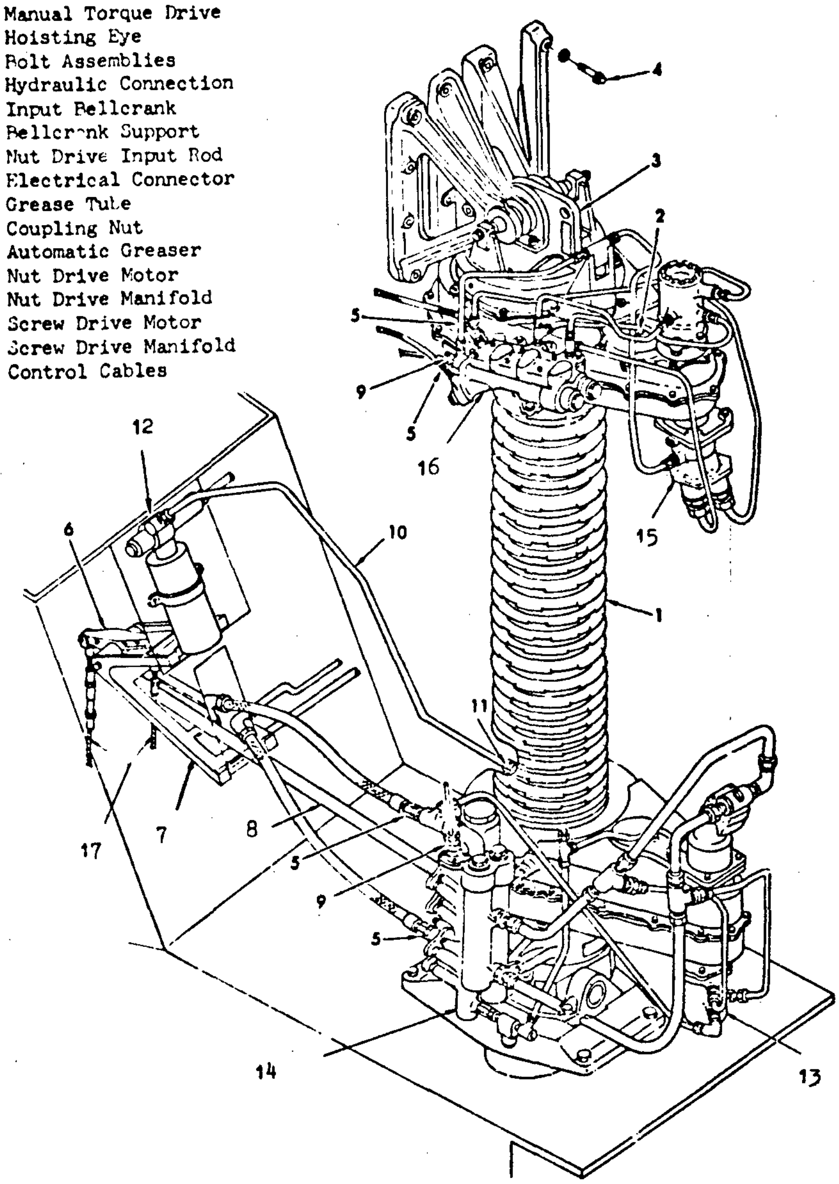


FIGURE NO. 1(3.2.6.2.1.1) C-5A PITCH TRIM ACTUATOR

Requirement

3.2.6.2.1.2 Ballscrews. An adequate number of balls and ball circuits shall be provided to keep individual ball loading within allowable nonbrinelling limits. On units used in essential and flight phase essential applications, at least two separate independent ball circuits and a secondary load path with load capability in accordance with 3.1.11.1.2 shall be incorporated.

Comparison

Ballscrews are utilized only in the C-5A leading edge slat system and the trailing edge flap system. Two separate and independent ball circuits are utilized in each ballscrew nut. The number of balls used in each circuit exceeds the minimum number required to prevent brinelling. Each ballscrew nut also includes a redundant load path in addition to the ball circuits, both of which meet the requirements of paragraph 3.1.11.1.2. With the loss of all the balls, the low efficiency of the redundant load path causes torque limiters in the affected system to trip and stop system operation. Because the system will not operate, the redundant load path is not exposed to operational fatigue loads.

In summary, the ballscrews used in C-5A secondary control systems are in compliance with this requirement.

Discussion

The requirement covers the subject addressed except for one aspect. There have been ballscrews used on aircraft that utilize two separate independent ball circuits, but which had a common retention device for holding the ball return tubes in place. Even though two separate ball circuits were used, this redundancy was nullified by the use of a common single retention device for the ball return tubes. The loss of one fastener or fracture of the retainer would result in the loss of all the balls. Obviously then, two separate retention devices or a single fail safe retainer with dual attachments must be used to maintain the integrity of the two separate ball circuits.

Recommendation

Add the following sentence to the end of the existing requirement of paragraph 3.2.6.2.1.2:

> "Completely separate and independent retainers or a single fail safe retainer with dual attachments shall be used for the ball return tubes or conduits."

Requirement

3.2.6.3 Mechanical Torque Transmitting Actuation. Specific approval from the procuring activity must be obtained before use of such provisions in essential and flight phase essential applications. Backlash accumulation shall not prevent the system from performing its required function throughout the service life of the airplane.

3.2.6.3.1 Torque Tube Systems. Torque tubes which are exposed to possible misuse, such as support for maintenance personnel, shall be shielded from such misuse or shall be of adequate stiffness to prevent damage to the installation. Each torque tube, in a linked run of tubes shall be removable and reinstallable in the aircraft without disturbing the support, component, or other interfacing system element at either end of the torque tube. Guards which are capable of containing a broken torque tube against thrashing shall be installed in appropriate locations to prevent damage to wiring, tubing, and other equipment. The rated operating speed of a torque tube system shall be no greater than 75 percent of the critical speed.

3.2.6.3.1.1 Torque Tubes. Torque tubes shall have a minimum wall thickness of 0.035 inch and shall be seamless, except that steel tubes, seam welded by the electrical resistance method, may be used.

3.2.6.3.1.2 Universal Joints. Univer al joints shall be in accordance with MIL-J-6193 or MIL-U-3963, as specified in AFSC Design Handbook DH 1-2, General Design Factors, Section 4C, Universal Joints, and shall not be used for angularities greater than specified therein or recommended for the specific component by the manufacturer.

3.2.6.3.1.3 Slip Joints. Adequate engagement shall be provided to insure that disengagement will not occur under all expected operating conditions, or due to buildup of adverse manufacturing and installation tolerances.

Comparison

Torque tube systems are used in the C-5A leading edge slat drive system and trailing edge flap drive system as shown in Figure No. 3 (3.2.3.1.1). The torque tubes vary in length, overall diameter and wall thickness; but in no case is a tube wall thickness less than 0.065 inch. Where torque tubes are exposed to possible misuse, they are protected by their inherent stiffness or by shielding. Each of the tube assemblies can be removed and installed without disturbing the supports or other interfacing system elements at either end of the torque tube. Guards which are capable of containing a broken torque tube against thrashing are used to prevent damage to wiring, tubing, and other equipment. The rated operating speed of the torque tube systems is less than 75 percent of the lowest critical torque tube speed.

The torque tubes are seamless aluminum and have a slip joint at one end of each tube. The slip joints are designed such that, under the most adverse

wing bending condition combined with the most adverse buildup of manufacturing and installation tolerances, the minimum engagement of the slip joints is one diameter. Finally, the hooks type universal joints used in the torque tube systems are in accordance with MIL-J-6193.

Discussion

There are two aspects of the requirement that need revising. One concerns the wording used in the second sentence of Paragraph 3.2.6.3.1. It is unclear whether the word "component" refers to another component of the torque tube system or to a component of another system. Assuming the former, the word "component" is redundant since the phrase "or other interfacing system element" means the same thing. However, there are occasions when it is necessary to include a torque tube as part of a subassembly that must be removed or installed as a unit because of configuration and/or installation constraints. The use of constant velocity joints instead of angle gearboxes in a torque tube system, for example, will sometimes cause this kind of situation.

The other comment concerns Paragraph 3.2.6.3.1.3. "Adequate engagement" does not provide sufficient direction to the inexperienced design engineer. Our experience at Lockheed indicates that an engagement of one diameter should be the minimum.

Except for the two comments stated above, the requirements are well stated. These requirements are definitely needed to provide the design engineer with the proper guidance in designing a torque tube system.

Recommendation

Revise the second sentence of Paragraph 3.2.6.3.1 as follows:

> "Unless otherwise approved by the procuring activity, each torque tube in a linked run of tubes shall be removable and reinstallable in the aircraft without disturbing the support or other interfacing system or element at either end of the torque tubes."

Revise Paragraph 3.2.6.3.1.3 as follows:

> "Slip joint engagement shall be sufficient to permit an engagement of one diameter as a minimum under maximum structural deflection in combination with the most adverse buildup of manufacturing and installation tolerances."

Requirement

3.2.6.3.2 Gearing. All gear boxes used in actuating systems shall meet the requirements specified in MIL-G-6641.

Comparison

The C-5A leading edge slat and trailing edge flap drive systems as well as the horizontal stabilizer pitch trim actuator assembly utilize angle, tee and hydraulic motor driven gearboxes. Although these gearboxes are in compliance with a few of the requirements of MIL-G-6641, this specification relates primarily to accessory drive gearboxes, as its title indicates.

In summary, the C-5A flight control systems are not in complete compliance with this requirement.

Discussion

MIL-G-6641 is a specialized document that has little applicability to the kind of gearboxes used in secondary flight control systems. The very few requirements, such as Paragraphs 3.1 through 3.6, of this specification that are valid for flight control gearboxes are covered by other specifications and are included in the CEI specification document.

Recommendation

It is recommended that this requirement be deleted.

Requirement

3.2.6.3.3 Flexible Shafting. Not applicable.

Requirement

3.2.6.3.4 Helical Splines. Involute helical splines shall use only the ASA standard tooth forms Numbers 1 through 5. Ballsplines shall meet the requirements specified in 3.2.6.2.1.2 for ballscrews.

Comparison

This requirement does not apply since no helical splines (Yankee screwdrivers) were used on the C-5A FCS.

Discussion

This is a valid requirement which does not apply to the C-5A FCS; it can be demonstrated; and it should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.6.3.5 Rotary Mechanical Actuators. Rotary mechanical actuators used with a through shaft which attaches to torque tubes at both ends, thus serving as a portion of the torque distribution system, shall be capable of reacting full system torque in both the forward direction (due to a jam anywhere in the system) or in the backdriving direction (due to overrunning load), unless provided with a torque limiter and no-back brake or other devices which would preclude such loading.

Comparison

The C-5A trailing edge flap and leading edge slat systems shown in Figure II-8 are driven by a common flap power package (FPP). The flap power package shown in Figure 1 (3.2.6.3.5) utilizes a thru shaft attaching to torque tubes at both sides thus supplying power to the torque distribution system which drives the panel actuators.

The output at the FPP is capable of developing the maximum output torque capability. To protect the system from a jam in the leading edge slat system or an excessive backdriving load, a decoupler is provided in the FPP torque tube system. This is supported by torque limiters and a leading edge slat brake assembly. Each ball screw actuator has an integral torque tube limiter to protect against an overload condition.

Discussion

The C-5A flap actuation system design meets the load protection requirements of this specification and can be readily demonstrated.

The requirement should be reviewed for clarification before being specified for future transport type aircraft.

There is some concern whether it was the intent of this specification to apply specifically to a "power hinge" application whereby the torque output of the actuator was to apply directly to the torque distribution either "ganged," "parallel," or a single unit and applied at the hinge line of the surface or component being actuated, or whether the intent was to be broad enough to cover any torque producing output system to supply power to a torque tube system regardless of the output force mode at the surface or component being actuated. The latter, broader interpretation was used to apply to the C-5A flap system design.

Even if one interprets this specification to apply to "power hinges" the reference to the use of torque tubes may preclude applicability to some types of power hinges which do not employ a torque tube system. For example. there have been proposals to use hydraulic rotary actuators either "ganged" or as single units which may be mounted to have their output torque coincident with the surface or component certerline of action.

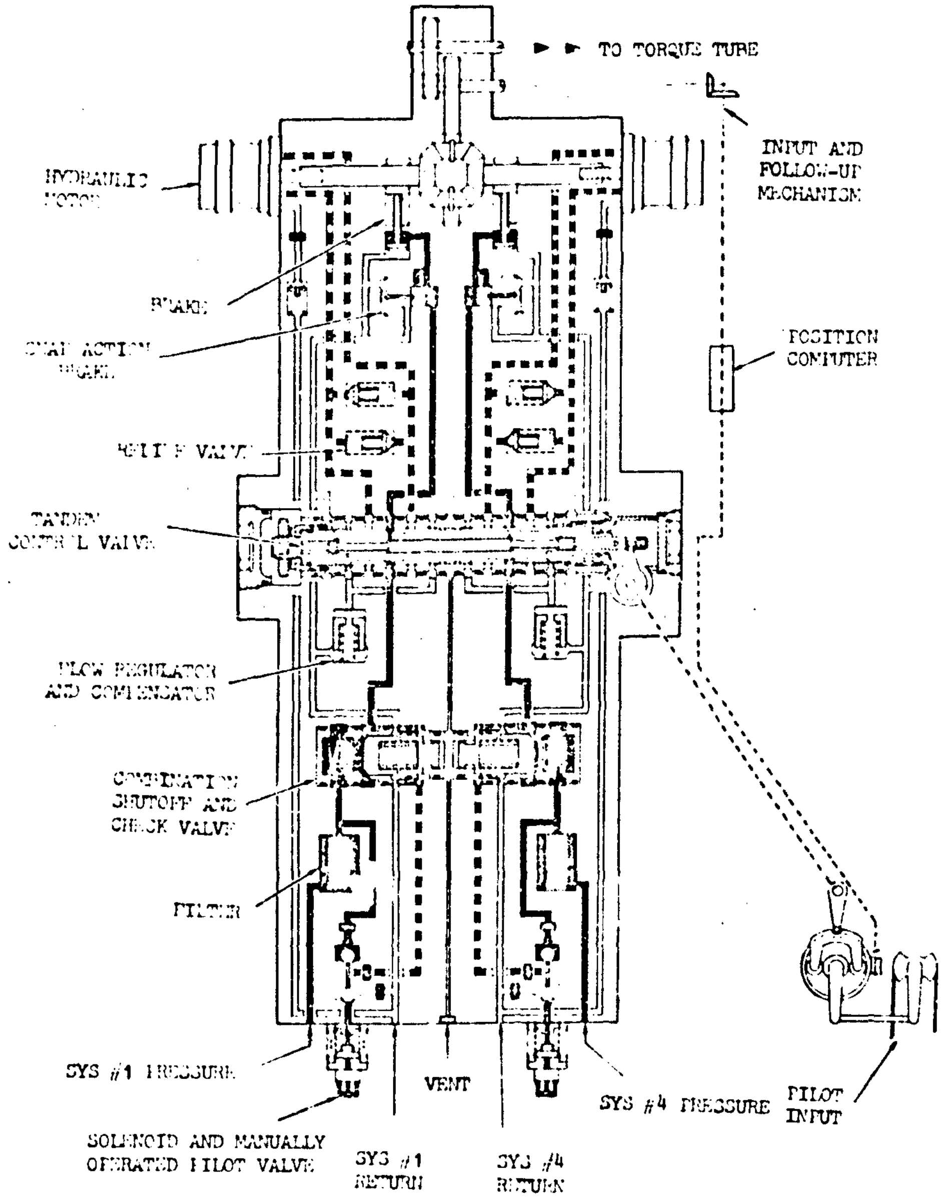


FIGURE NO. 1(3.2.6.3.5) C-5A FLAP POWER PACKAGE

It is recommended that the intent of this requirement be reviewed for a more general applicability for all rotary actuation systems such that advancements in the state-of-the-art may be fully utilized for future applications.

Recommendation

It is recommended that a more general specification wording be employed. Change the title and wording as follows:

> "3.2.6.3.5 Rotary Actuators. Rotary actuators which provide a rotary output torque and which either form a portion of the torque distribution system or provide the total output torque as a single unit, shall be capable of reacting full system torque in both the forward direction (due to a jam anywhere in the system) or in the backdriving direction (due to an overrunning load or jam), unless provided with a torque limiter and no-back brake or other devices which would preclude such loading."

Requirement

3.2.6.3.6 Torque Limiters. Where used, torque limiters designed to slip or lock to adjacent structure shall be properly located in the transmission system to prevent drive loads in excess of control surface limit load from being transmitted past the limiter(s) and the spring rate of the transmission system shall be matched so that the stress in any member due to sudden application does not exceed its yield strength.

Comparison

The C-5A utilizes torque limiters only in the leading edge slat drive system and the trailing edge flap drive system. Each ball screw actuator in both systems incorporates a torque limiter. However, the leading edge slat drive torque tube system in each wing semi-span also incorporates three torque limiters as shown in Figure No. 3 (3.2.3.1.1).

These torque limiters are bi-directional in that they limit the torque applied to the system by the trailing edge flaps power package in addition to limiting the feedback torque due to aiding loads in excess of the trip point. More specifically, the torque tube drive system torque limiters limit the maximum driving load, including "leak-through" torque, in the system outboard of the torque limiters. The load produced in that part of the system shall not cause yielding when the system is subjected to the impact loads. Impact loads can be produced by a slat ballscrew actuator being misrigged to cause its non-jamming stop to be engaged while the system is running at full rated speed in the retract or extend direction with an aiding airload applied to the system. In addition, the torque limiters shall not trip during slat extension or retraction with the sudden application of limit air loads. Finally, if one of these torque limiters trips due to a jam, the maximum "leak through" torque of the torque limiter shall not be sufficient to cause permanent set in the system outboard of the torque limiters or cause its performance to fall outside of specification requirements. The trip torque of the torque limiters is set at an opposing load sufficiently above limit air load over a temperature range of -65° F to +160° F to preclude nuisance tripping.

Each slat actuator contains a torque limiter that limits the applied torque from the input shaft of the actuator gearbox to the ballscrew. This torque limiter is also bi-directional in that it is capable of limiting feedback torque during high speed flights caused by airloads applied at the forward end of the actuator in excess of the trip point. The trip setting of the torque limiters is such that they will not trip during extension or retraction with the sudden application of limit loads over a temperature range of -65°F to +160°F. If a slat actuator torque limiter trips due to a jam, the maximum "leak-through" torque and the column load it produces at the ballscrew must not cause permanent set in any part of the actuator assembly.

The torque limiters used in the trailing edge flap actuators function similarly to those in the leading edge slat actuators. However, they do not have the capability of absorbing feedback torque due to excessive

aiding airloads, which could occur during retraction, without tripping. The trip setting is based on maximum normal operating loads rather than limit loads as with the leading edge slat system torque limiters. The trailing edge flap actuator torque limiters are set to trip at an opposing load sufficiently above maximum normal operating loads over a temperature range of -65° F to +160° F to preclude nuisance tripping and prevent the generation of loads in excess of limit load. The C-5A meets and exceeds the intent of the requirement.

Discussion

The requirement is well stated and is applicable to large transport aircraft.

Recommendation

Accept "as is."

Requirement

3.2.6.3.7 No-Back Brakes. No-back brakes shall prevent back driving (or feedback) forces imposed on the output of an actuating mechanism from being converted to torques which can cause the input shaft to rotate. In no-back brakes of the heat dissipative type, provisions shall be included to distribute heat generated by the brake so that temperature limitations are not exceeded.

Comparison

The C-5A utilizes no-back brakes only in the pitch trim actuator assembly shown in Figure No. 1 (3.2.6.2.1.1). Two no-back brakes are used, one in the nut drive located at the bottom of the assembly and one in the screw drive located at the top of the assembly. The no-back brake is hydraulically disengaged (released) when hydraulic pressure is applied to the drive motor. Each brake utilizes a stack of brake disks which have the capability of dissipating heat generated during operation. Removal of hydraulic pressure from the drive motor results in the no-back brake being automatically engaged. Engagement of the no-back brake prevents airloads imposed on the actuator from feeding back, causing a change in trim setting. The no-back brakes used in the C-5A pitch trim actuator are therefore in compliance with the intent of this requirement.

Discussion

The requirement is reasonable and complete except for additional clarity on one point. The requirement "provisions shall be included to distribute heat generated by the brake so that temperature limitations are not exceeded" needs to be discussed in more detail in the "Users Guide." It is suggested that the following information may be appropriate. "No-back brakes shall prevent back driving (or feedback) forces imposed on the output of an actuating mechanism from being converted to torques which can cause the brake input shaft to rotate. In no-back brakes of the heat dissipative type, provisions shall be included to insure that the heat rejection of the brake, while operating under rated duty cycle conditions, is sufficient to prevent the design limit operating temperature from being exceeded."

Recommendation

Accept "as is."

Additional Data

The sentences indicated by the left vertical sideline should be added to the background information and "Users' Guide."

Requirement

3.2.6.4 Hydraulic Actuation. Hydraulic actuation components shall be designed in accordance with MIL-H-8775 or MIL-H-8890 and specific component specifications as applicable. If hydraulic bypass provisions are necessary to prevent fluid lock or excessive friction load or damping, bypassing and resetting shall occur automatically when system pressure drops below or returns to the minimum acceptable value for actuation. In actuation systems designed for manual control following hydraulic failure, provisions shall be made to permit bypassing of the hydraulic systems for checkout purposes and to permit pilot training with the emergency manual system.

Comparison

The C-5A flight control system (FCS) hydraulic actuation components were designed in accordance with the requirements of Contract End Item (CEI) specification CP40002-6B, MIL-H-8775, and the component specification requirements. All actuation systems were designed to MIL-H-8775 using MIL-H-5440, Type I, 3000 psi systems and hydraulic fluid conforming to MIL-H-5606. Figure 1 (3.1.4), Item 9, shows a typical hydraulic system bypass and shut-off valve used in the C-5A-FCS primary servo actuator manifold. The normal bypass and shut-off valve functions are to shut off the hydraulic system pressure coming into the servo actuator manifold and to provide a relatively unrestricted bypassing of fluid between both sides of the actuation piston head. If a hydraulic system loses pressure and if the servo control valve is moved to accommodate normal pilot input motion, then hydraulic fluid bypassing automatically occurs through the main servo control valve orifices without the necessity of actuating the shut-off and bypass valves. There is no resultant excessive friction or damping. The normal bypass and shut-off function gives the pilot and ground crew the option of isolating and shutting off a certain hydraulic system at a particular servo actuator location for purposes of checking out each system operation at each servo actuator manifold. On Figure 1 (3.1.4), Item 10, is a gust lock check valve which acts as a control surface gust lock when the aircraft is parked and the hydraulic systems are de-energized. However, for the gust lock to function, the shut-off and bypass valve must be left in the normal "pressure on" position and the pilot controllers must be left free (hands off). In this application the hydraulic lock is necessary to function as a surface gust lock.

The C-5A rudder servo actuator utilizes a different shut-off and bypass valve than the other primary servo actuators. During normal operation with full system pressure on, for functional reasons, hydraulic fluid is allowed to bypass between the two sides of the actuation piston.

To provide for the rudder servo gust lock function, when hydraulic systems are de-energized, the bypass path of fluid between the two sides of the actuation piston is blocked off during the shut-off mode. During normal operation of the rudder servo with one hydraulic system turned off, hydraulic fluid bypassing, in the deactivated system, is provided by normal movement of the main control valve and subsequent bypassing of hydraulic fluid through the control valve orifices.

Discussion

The C-5A FCS meets the intent of the basic specification requirements. However, the requirement to provide automatic hydraulic system bypassing and resetting as a function of a pressure drop, when it is necessary to prevent a fluid lock or excessive friction or excessive damping, may be found in conflict with other requirements. The comparison of the C-5A FCS above points out a need for design flexibility as to when a hydraulic lock may be required. In other words, different design criteria or the actual evolution of the system development may require a different configuration. Even the "hydraulic lock" discussed in the C-5A FCS is controllable through the bypassing operation from the main control valve movement. There may be circumstances where automatic bypassing, which can not be controlled by the pilot, would be undesirable. On the other hand, the pilot should always be given the flexibility of providing a bypass function of the servo actuation system when an inadvertent hydraulic lock might occur due to a malfunction of the system. An example is an inadvertent stuck or neutrally centered control valve with the surface hard over and full system pressure on. In this case the automatic bypass would not function and even shutting off system pressure might not assure operation of the bypass and subsequent trailing of the surface.

The point is that there are many good applications of automatic bypassing, but not all are. In fact, there may be other better means, either hydraulically, mechanically, or electrically of providing a bypass function.

The imposition of a particularly detailed requirement, in a general specification, ignores many other similarly detailed design considerations which might also be imposed under the rules or guidelines existing at that time. However, because of the multitude of different configurations and criteria imposed on hydraulic actuation systems and because of constant advances in state-of-the-art technology, the contractor should be given the flexibility of providing the most cost effective and safest design.

Design guidelines which recommend this type of design consideration are good and their intent should be presented to the contracting agency.

It would be beneficial to develop and compile more specific requirement guidelines from the results of hardware "tradeoff" studies to present particular design practices versus function criteria, operation, cost, weight, vulnerability, etc., for future inclusion in the "Users Guide" or reference to an AFSC or similar source of information.

Recommendation

Revise the second sentence to read: The decision for automatic versus manual hydraulic bypass provisions shall be justified in the design process.

Additional Data (For Users Guide)

Bypass and Shut Off Provisions - There are worthy applications for hydraulic fluid bypassing (i.e. free flow of hydraulic fluid between both sides of the piston). Some applications may include preventing fluid lock and preventing excessive friction and/or damping. However, there are other legitimate design considerations in determining whether the by-passing should be automatic (as a function of criteria such as hydraulic pressure drop) or manual (controlled by the pilot) or possibly a combination of alternative control modes. Manual control of hydraulic fluid bypassing may be desirable for ground or in flight checkout or emergency control even for full power systems that don't have manual reversion. C-5A ground gust lock design features, for example, prohibit the use of automatic fluid by passing.

Requirement

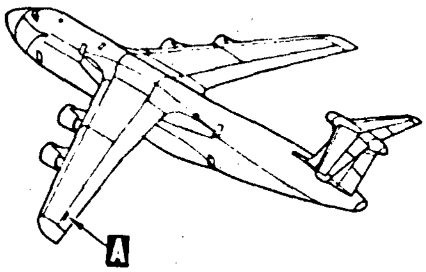
3.2.6.4.1 Hydraulic Servoactuators. Hydraulic servoactuators shall be designed in accordance with ARP 1281. Electrohydraulic servovalves shall be designed in accordance with MIL-V-27162. If electrical-input hydraulic servovalves having mechanical feedback of actuator position are used, the applicable requirements of ARP 988 shall be met.

Comparison

The C-5A primary FCS uses hydraulic servoactuators which were designed to meet the performance requirements of Contract End Item specification CP 40002-6B. The aileron, rudder and elevator surface servoactuation systems used an arrangement of separated manifold and actuator cylinders which were hydraulically connected by flexible connections. Figure No. 1 (3.1.4) is the inboard elevator servo hydraulic schematic and is typical of the primary hydraulic servoactuators which use the integrated stability augmentation system (SAS). This illustrates the typical dual hydraulic system irreversible servo which has parallel pilot and autopilot mechanical inputs and series stability augmentation electrical inputs that are translated into surface position thru the closed loop servo system. The actuator cylinders are the load carrying members. These are separately mounted to fittings on the primary structure beam and the actuating surface beam through self-aligning bearings. The hydraulically servoed extension or retraction of the cylinders provides the demanded surface displacement. All hydraulic seals for moving components are dual and are vented to return between the seals to permit leak detection and increase seal life. Each hydraulic manifold is rigidly attached to the aft face of the primary structure beam. The manifold consists of a tandem dual concentric servo main control valve (MCV), shut-off valves, pressure switches, relief valves, filters, and gust lock check valves. The outer sleeve of the MCV is actuated by the SAS tandem actuator. This actuator is controlled by duplicated sets of a single stage electrohydraulic valve, solenoid shut-off valve and elements of the closed loop SAS system. These elements include the linear variable differential transformer (LVDT). The mechanical input controls the inner spool of the MCV and a mechanical summing linkage provides the mechanical closed loop for the feedback system. Coiled tubing assemblies or rotary couplings (for aileron servo) ports hydraulic fluid from the manifold to each actuator cylinder. The outboard elevator mechanization differs in that the servos include no SAS and triplex actuation is provided by use of a single manifold and actuator in addition to the dual actuators and manifolds. The basic reasons for using the separated hydraulic components instead of the integrated manifold and tandem servo actuator were installation space limitations, cost and weight considerations, and LRU maintainability.

Figure No. 1 (3.2.6.4.1) shows the aileron servo actuator installation mounted on the wing box structure beam. Deep penetration of the wing box beam with massive structural beef-up would have been necessary for the integrated tandem actuator design. It can be seen that the separated

1. BOLT, WASHER, SAFETY WIRE (4 PLACES)
2. BOLT, NUT, WASHER, COTTER PIN
3. HYDRAULIC CONNECTIONS
4. BUNGEE ROD
5. MANIFOLD
6. BOLT, NUT, WASHER, COTTER PIN
7. FEEDBACK BUNGEE ASSEMBLY
8. ELECTRICAL CONNECTIONS (TYPICAL 12 PLACES)
9. BELLCRANK
10. LINK (CONNECTS TO AILERON FITTING)
11. ROTARY JOINT (2 PLACES)



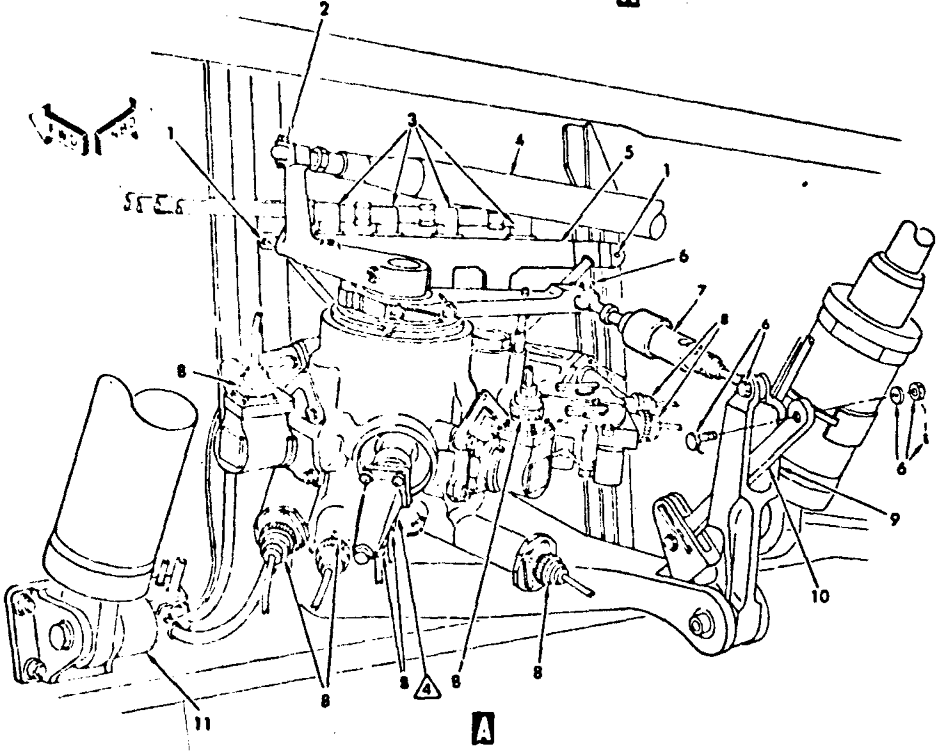


FIGURE 1 (3.2.6.4.1) AILERON SERVO INSTALLATION

actuator mounting helps to spread the actuator loading into the wing box and the actuator mounting brackets are backed up by the more rigid wing box structural caps. Items 9 and 10 of Figure No. 1 (3.2.6.4.1) show the idler bellcrank and feedback link (from the aileron surface) which provide the necessary feedback magnitude and direction to assure a stabilized servo loop throughout the operational frequency envelope.

Figure No. 2 (3.2.6.4.1) is a typical flight spoiler servo actuator installation. The flight spoiler servo is a dual tandem hydraulic system irreversible servo actuator which departed from the usual C-5A primary servo design because of the installation space requirements. Figure No. 3 (3.2.6.4.1) is the flight spoiler servo schematic which shows a mechanically controlled MCV, shut-off valves, pressure switches, relief valves, filters, gust lock check valves and actuation. The dual tandem piston actuation system is attached to structure thru self-aligning bearings. Figure No. II-9 indicates the arrangement of the ten flight spoiler servo actuators which are used both in flight for roll control and on the ground for lift spoiling.

The criteria for the C-5A servo actuator design were completely defined in the detail specification document for procurement from the subcontractor. Complete envelope drawings defined the mechanical installations and arrangements including all relevent interfaces such as actuator mountings, feedback arrangements, input arrangements, input and surface travels, etc.

The backup attachment structure spring rate requirements and actuated surface inertias were specified in the detail specification document. The servo loop stiffness requirements considered both static and dynamic criteria. The static loop stiffness requirements used a minimum hydraulic fluid bulk modulus and were derived from aeroelastic characteristics, servo to surface gearing (gain) and surface deflections as a function of the aerodynamic hinge moment. The dynamic stability was defined by bode plot parameters which, for the aileron servo, had a closed loop response upper amplitude ratio limit of +2 db and lower limit of -5 db out to 11 radian/second. The lower phase limit curve was non-linear from a point at 1 radian/second of 25° up to 125° at 30 radians/second. Input required sweeps of ±1° over 3 to 300 radians/second and ±3 1/2° over 3 to 6 radians/second. During the evolution of the primary FCS servoactuators the stability criteria were met in some areas by supplemental stabilizing techniques such as stabilizing mechanical feedback links, increased structural stiffness and hydraulic leakage.

The electrohydraulic (E-H) control valves were designed to MIL-V-27162 basic (Flow Control) except as noted in the detail component specifications. The E-H valves, as noted, are used to control the SAS actuation piston for the MCV which uses the LVDT electrical feedback. The servoactuators were designed to the hydraulic systems requirements of MIL-H-5540, Type I, and hydraulic component requirements of MIL-H-8775. The actuating cylinders

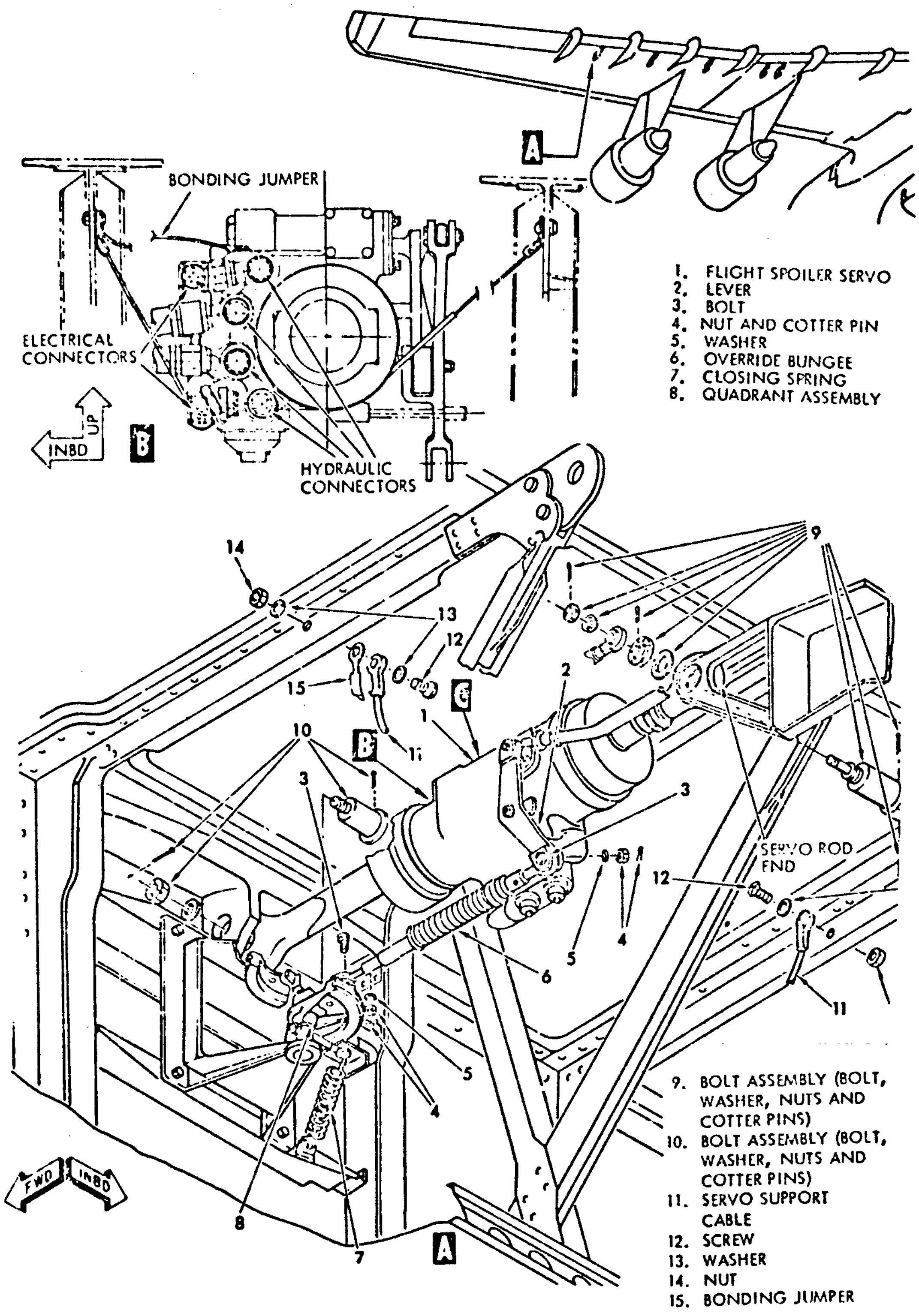


FIGURE NO. 2 (3.2.6.4.1) FLIGHT SPOILER SERVO INSTALLATION

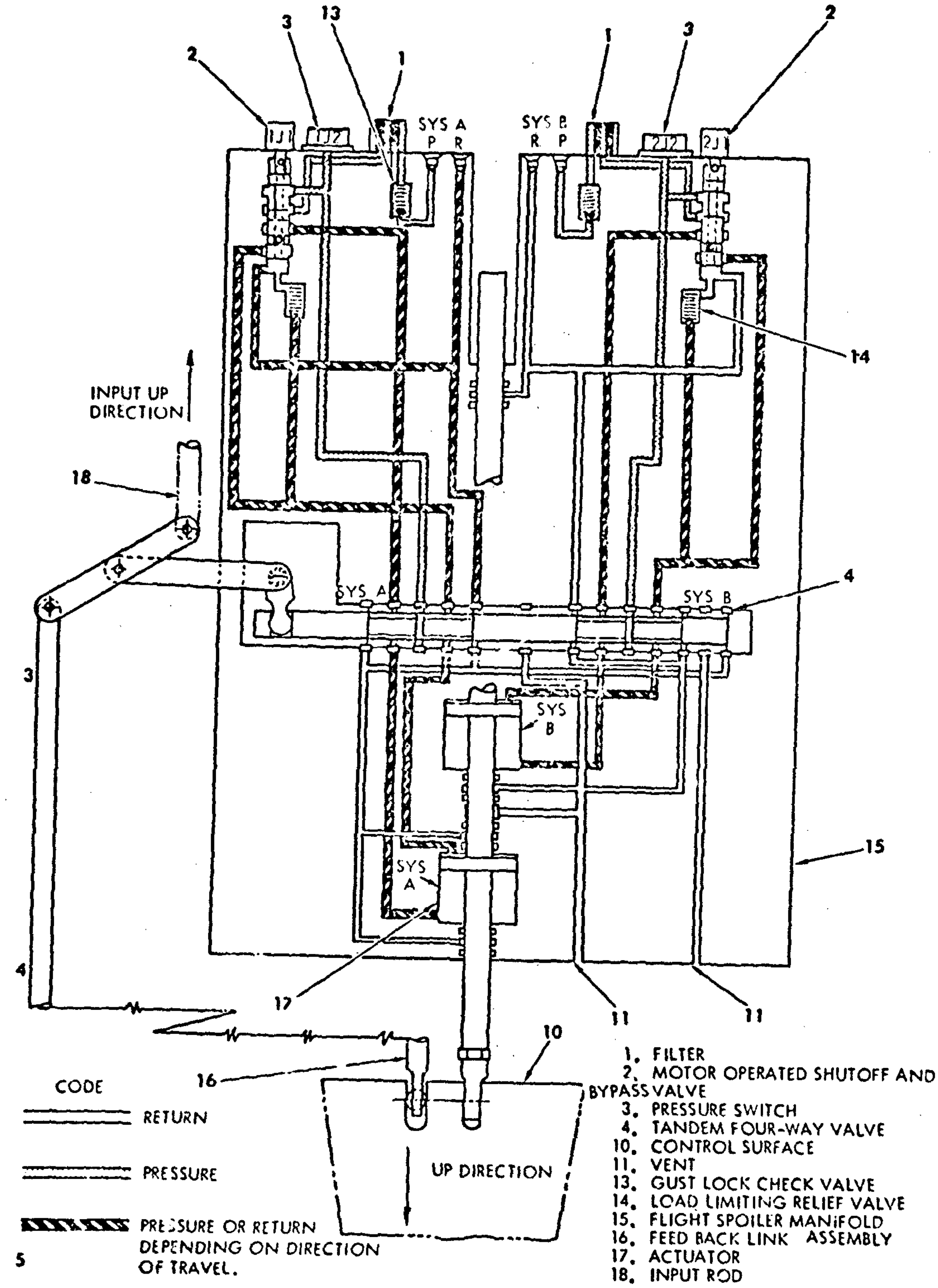


FIGURE NO. 3 (3.2.6.4.1) FLIGHT SPOILER SERVO SCHEMATIC

were designed to MIL-C-5503, Type I. Exceptions to these military specifications were noted in the detail specifications. The endurance life of the primary FCS servoactuators was more than 5,000,000 cycles as noted in the validation for Paragraphs 3.1.11.3 and 3.1.12. The motor or solenoid operated shut-off and bypass valves (MSBV) were designed to the requirements of MIL-V-7915 with the electromechanical actuators per MIL-A-8064 and solenoids per MIL-S-4040, except as noted in the detail component specification. Hydraulic filters were designed to MIL-F-8815 and check valves were designed to MIL-V-5524. The servoactuator assembly and components were required to meet the environmental test requirements of MIL-STD-810. The detail component specification listed all the basic performance criteria such as velocity, threshold, hysteresis, leakage, strength, output capability, dynamic synchronization, null shift, etc., required to meet all the FCS functional, safety, reliability, and maintainability requirements.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Additional Data (For Users Guide)

Actuator Mechanization - The 'users guide' accompanying the MIL-A-XXXX military specification: 'Aircraft Flight Controls, Power Operated, Hydraulic General Specification For' which replaces ARP 1281 should be consulted regarding detail design considerations for hydraulic servoactuators.

Requirement

3.2.6.4.2 Motor-pump - Servoactuator (MPS) Package. This is defined as an integrated servoactuator package which incorporates an electric motor driving a hydraulic pump, a hydraulic fluid reservoir, a servoactuator, and necessary accessories packaged in a single, self-contained LRU. Individual components within the integrated package shall be designed in accordance with the applicable requirements of the corresponding component specifications. Essential or flight phase essential applications require specific approval from the procuring activity.

Comparison

The C-5A flight control systems do not utilize motor-pump - servoactuator packages.

Discussion

The requirement addresses the components of an MPS package and ignores the package as a whole. As a minimum, it should require an overall specification document be prepared of which the individual component specification documents become a part. The overall MPS package specification document should address the package/control and power systems interface as well as the topics normally covered by a specification document for an ordinary control surface power servo-actuator assembly.

Recommendation

1. It is recommended that the following sentence be added between the existing first and second sentences.

   "In addition to the topics normally covered, the MPS package design specification document shall address the functional, performance and interface requirements of the individual components that make up the MPS package."

2. It is recommended that the existing second sentence be revised as follows:

   "Individual components within the integrated package shall be designed in accordance with the applicable requirements of the corresponding components specifications which shall be referenced in and be a part of the MPS package specification document."

Requirement

3.2.6.4.3 Actuating Cylinders. Actuating cylinders without control valves and feedback provisions in the same LRU shall be designed in accordance with MIL-C-5503, except that the life cycling requirements shall be modified to reflect the specific usage. (See 3.1.12.)

Comparison

The C-5A FCS primary hydraulic servo actuator cylinders are separate LRU components of the total servo installation. Typical examples of C-5A servo actuators are shown for the elevators in Figures No. 1 (3.1.4) and No. II-5 and for the ailerons in Figure No. 2 (3.2.3.2.5). The actuating cylinders were designed to meet or exceed the minimum requirements of MIL-C-5503. In some areas C-5A design requirements were more stringent. The C-5A FCS primary hydraulic servoactuators were designed and endurance life cycle tested to a minimum of 5,000,000 cycles of various load and stroke combinations consisting of anticipated amplitudes, frequencies and loads related to the cyclic distribution defined in MIL-C-5503C. They met the design goal by achieving a seal life equal to the actuator endurance life requirement. See Paragraph 3.1.12 validation for additional discussions relating to wear life.

Discussion

The wording of the subject specification does not rule out the application of actuating cylinders in servoactuator applications as noted in the referenced C-5A servo actuator design. The actuator component is part of a servo loop installation. However, the actuator is a separate LRU and does not contain a control valve or feedback provision. The C-5A flight spoiler actuator is a tandem servo actuator with an integral control valve and feedback provision. Some of the guidelines of MIL-C-5503 were applied to the flight spoiler servo actuator as design criteria. MIL-C-5503 provides good design criteria; however, as noted in the "Users Guide," it should not be the limiting criteria for designing a long life servo actuator. This paragraph of MIL-F-9490D could impose MIL-C-5503 as the sole servo actuator design criteria, except for life cycle requirements. Therefore, this requirement should be reworded to allow more flexibility in deviating from MIL-C-5503 when its criteria would restrict the contractor from meeting a demanding servoactuator requirement.

With the recommended modification this is a good requirement which has been satisfied by the C-5A control system design. It can be readily demonstrated. and should be specified for all future transport type aircraft.

Recommendation

Revise the wording of 3.2.6.4.3 as follows:

"Actuating cylinders without control valves and feedback provisions

in the same LRU shall be designed in accordance with MIL-C-5503, except where more demanding design criteria may be required to assure meeting overall system life and/or aircraft mission reliability requirements.

Additional Data (For Users Guide)

Actuating cylinders without control valves and feedback provisions in the same LRU which are part of a total servo loop installation should be designed to comply with all of the applicable servoactuator design requirements. In addition to more stringent life cycling requirements other considerations include stiffness and servo system compliance. An example of this type of servoactuator is the C-5A Aileron Servo Actuator installation shown in Figure [No. 1(3.2.6.1.1)].

This design installation was chosen over the "Integrated" servoactuator design as a result of cost and weight trade studies. NOTE: For Figure [ ] use the next available users guide number.

Requirement

3.2.6.4.4 Force Synchronization of Multiple Connected Hydraulic Servo-actuators. In essential and flight phase essential flight control actuator installations employing multiple connected servoactuators, the actuators shall be synchronized as necessary to assure specified performance and durability as specified in 3.1.11.3 in the structure between actuators without undue structural weight penalties.

Comparison

C-5 servo actuator specifications provide for valve static and dynamic synchronization over the entire range of servo valve displacement. Synchronization between the two hydraulic systems is obtained by dimensional control of the servo valve spool and sleeve to control load differences between the actuators. In the case of the outboard elevator a dual servo and a single servo are used. The dual servo is rigged first, as above, with the single servo deactivated. The single servo input is then rigged unpressurized to rig pin locations. Finally the single servo output feedback rod is rigged to achieve approximately equal pressures on each side of its actuator piston (as sensed by a differential pressure sensing device which is contained in the servo manifold).

Discussion

Lockheed has met the requirements for force synchronization. The servo and rigging provisions have assured that the surfaces, actuators, and support structure are not subjected to significant out-of-phase loading conditions which might degrade structural or actuator life or increase structural weights. The requirement is realistic and desirable for future transport aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.6.4.5 Hydraulic Motors. Hydraulic motors may be used to actuate relatively low-duty-cycle, noncritical flight control surfaces, such as wing flaps, but specific approval from the procuring activity must be obtained before use in high duty cycle noncritical applications or in any essential or flight phase essential application. They shall be designed in accordance with MIL-M-7997.

Comparison

The C-5A utilizes hydraulic motors only in low-duty-cycle flight control systems. Specifically, two motors are used in the trailing edge flap drive power package assembly and two are used in the pitch trim actuator assembly. The two motors used in the flap system are the same, but the two motors used in the pitch trim system are different in displacement. All these motors are designed in accordance with MIL-M-7997. The C-5A is in compliance with this requirement.

Discussion

Lockheed considers this to be a good requirement applicable to future transport aircraft.

Recommendation

Accept "as is."

Requirement

3.2.6.5 Electromechanical Actuation. Electric power may be used to actuate relatively low-duty-cycle, noncritical flight control functions, such as for trim and in the AFCS, but specific approval from the procuring activity must be obtained before use in essential and flight phase essential applications. Electromechanical actuation components shall be designed in accordance with MIL-E-7080, and specific component specifications as applicable, and the following. Performance requirements shall be adequate for intended application.

Comparison

The C-5A FCS uses electromechanical actuators for the roll trim control, spoiler/aileron mix, ratio changer, rudder trim, rudder limiter, cross-wind steering, hydraulic servo shut-off valves, and in the throttle control console. These electromechanical actuators were designed to MIL-A-8064 for electromechanical actuators with electric motors per MIL-F-7969 and limit switches per MIL-S-6743. MIL-E-7080, Electric Equipment Installation and Selection, was also applicable.

Figure No. 1 (3.2.6.5) shows an exploded view of a trim actuator which is representative of the typical linear electromechanical actuator used in the C-5A FCS. From this figure it can be seen that the predominate portion of the actuator consists of mechanical components, such as drive gears, shafts, bearings, stops, etc. The primary specification for the design and testing of the electromechanical actuators was the component specification, using the requirements of MIL-A-8064, unless otherwise noted in the component specification.

Discussion

The C-5A FCS electromechanical actuators were designed to meet the electric equipment installation requirements of MIL-E-7080 in addition to other applicable specifications. However, MIL-E-7080 only covers the electrical equipment design and does not provide any guidelines for the mechanical portion of the design, such as that provided by MIL-A-8064.

It is recommended that additional design guidelines be provided both in the "Users Guide" and the subject specification to cover the mechanical design requirements. This requirement can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Rewrite the last two sentences to read:

> "Electromechanical actuation components shall be designed in accordance with MIL-E-7080, MIL-A-8064, and the specific component specification as applicable."

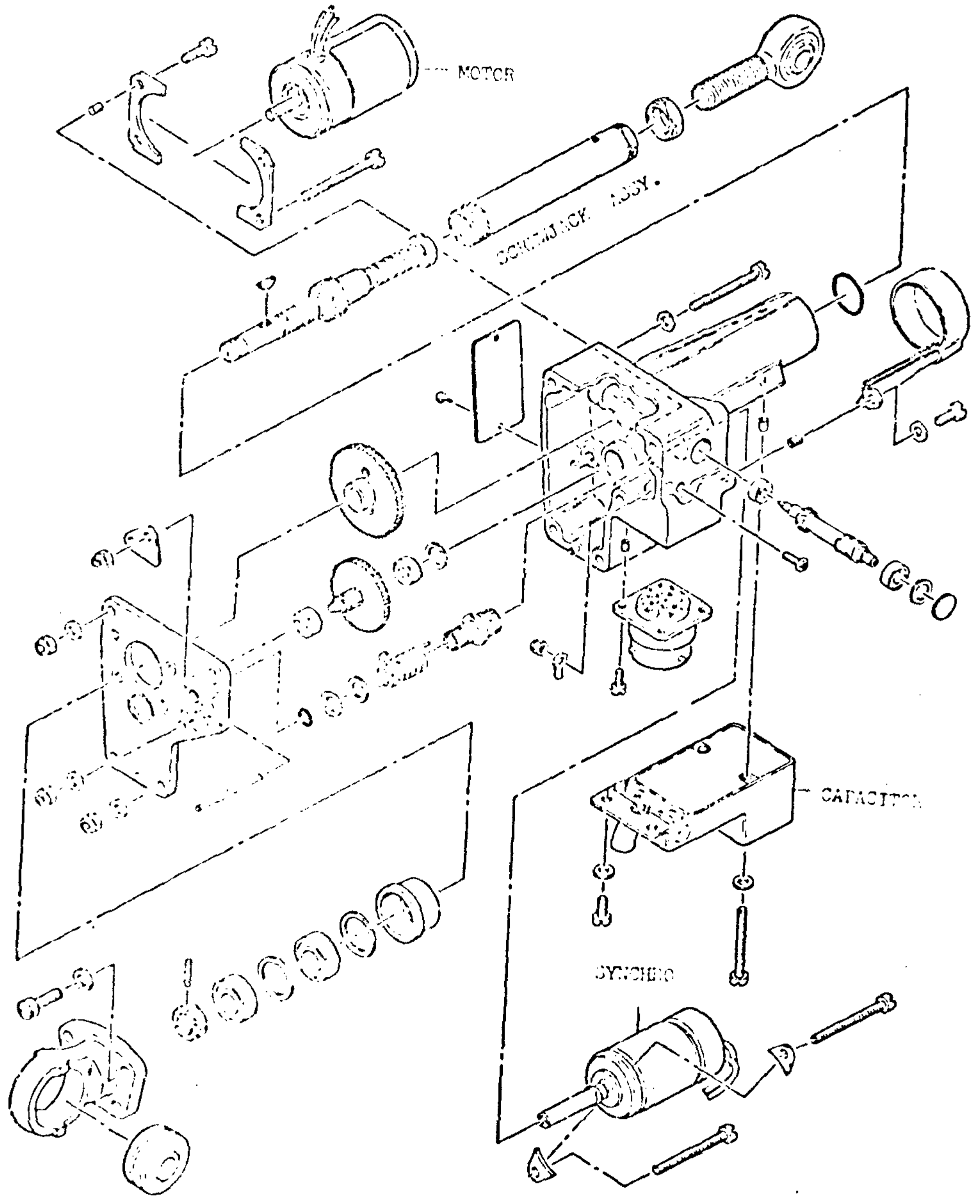


FIGURE NO. 1 (3.2.6.5) LINEAR ELECTROMECHANICAL ACTUATOR

Requirements

3.2.6.5 Electromechanical Actuation.

Additional Data

Linear electromechanical screwjack actuators, such as the one shown in Figure [No. 1 (3.2.6.5)] , are used for many non-critical flight control functions. The C-5A heavy logistics transport roll control system, Figure [No. II-4] illustrates typical applications of electromechanical trim actuators. Roll axis trim is accomplished by providing a series input to the aileron servo actuator. The Spoiler/Aileron Mix Box in the roll control input system provides sequencing and ratio changing functions by operation of linkages actuated by an electromechanical actuator.

Note: For Figure [No.] use the next available "Users' Guide" number.

Requirements - Not Applicable

3.2.6.6 Pneumatic Actuation

3.2.6.6.1 High-pressure Pneumatic Actuation

3.2.6.6.2 Pneumatic Drive Turbines

Requirement

3.2.6.7 Interfaces Between Actuation Systems, Support Structure, and Control Surfaces

3.2.6.7.1 Control Surface Stops. Surface stops shall be provided each flight control surface to positively limit its range of motion. Stops shall be located so that wear, slackness, or takeup adjustments will not adversely affect the control characteristics of the airplane because of a change in the range of surface travel. Each stop shall be able to withstand any loads corresponding to the design conditions for the control system. Where power control actuators are attached directly to the control surface, stops shall be provided within the actuator. Such actuators shall not only be designed for maximum impact loads, but also for the cumulative fatigue damage due to load cycling predicted during flight and due to bottoming during ground checkout and taxiing. Where control valve command input stops are provided, the actuators shall be designed for maximum impact stop loads, and not for fatigue damage due to bottoming, except as normally encountered with the input stops and feedback provisions functioning.

3.2.6.7.1.1 Adjustable Stops. All adjustable stops shall be positively lockhed or safety wired in the adjusted position. Jam nuts (plain or self-locking type) are not considered adequate as locking devices for this application.

Comparison

C-5A control surface travels are limited by the surface actuators which are attached directly to the control surfaces. The actuators have been designed to withstand the impact loads which occur during system no-load ground operation. In no case is a surface travel limit set by valve command input stops. Except for the spoiler actuators, all surface actuator lengths are established at the factory and are locked sealed and not meant to be changed in the field. The spoiler actuator lengths are adjusted when an actuator is installed initially or replaced. These adjustments are provided by threaded and slotted rod ends engaging the actuator piston rods. Jam nuts are used in combination with NAS1193 locking devices and safety wire is installed to prevent loosening of the jam nuts.

All surface stops are mechanical (pistons bottom in the cylinder body); are immersed in hydraulic fluid and are not subject to wear. All actuators were designed and tested to demonstrate their capability to withstand the cumulative fatigue damage due to load cycling predicted during flight and due to bottoming during ground checkout and taxiing. For instance, all flight control actuator (except ground spoiler actuators) were subjected to the following cycling during C-5A qualification tests:

- 2000 slam endurance at maximum rates, equivalent system mass moments of inertia and structural spring rates

- 5,000,000 endurance cycles including at least 4,980,000 load cycles and 20,000 unloaded cycles during which the actuators were bottomed at the extremes with full system pressure differential.

Ground spoiler actuators were subjected to 20,000 slam endurance cycles at maximum rates with equivalent system mass moments of inertia and structural spring rates. They were also subjected to 20,000 load stroke cycles with full system pressure being developed at the extend and retract positions. An additional 20,000 load cycles simulating spoiler load reversals due to flap operation with the spoilers closed and locked were also conducted.

Discussion

The requirements are valid and the C-5A actuators met or exceeded them.

Recommendation

Accept the requirements as they stand.

Requirement

3.2.6.7.2 Control surface ground gust protection. All flight control surfaces shall have provisions to prevent damage from ground wind loads as specified in MIL-A-8865. However, no separate provisions are required if the damping characteristics of installed flight control actuators suffice for gust protection.

Comparison

The C-5 flight control surfaces and structure were protected from damage from ground wind loads as specified in MIL-A-8865 by the surface actuator themselves. All flight control actuators except ground spoiler actuators utilize hydraulic snubbers integrated with the hydraulic servo valve and manifold. The manifold incorporates a gust lock check valve in the inlet pressure passage which prevents reverse flow of hydraulic fluid from the manifold to assure the presence of hydraulic fluid in the surface actuator when the servo is in the shut off and by-pass position. Figure 1 (3.1.4) Hydraulic Schematic inboard Elevator Servo is representative of all the flight control actuator snubbing.

Ground spoiler actuators incorporate integral mechanical devices in the actuators which lock the surfaces in the faired position. This lock is released only when hydraulic pressure is ported to the opening side of the actuator piston. Therefore, it serves as a ground gust lock to satisfy the requirements. Figure 1 (3.2.6.7.2) shows the ground spoiler actuator hydraulic schematic.

Discussion

The requirement is satisfied by the C-5A flight control system. The requirement is reasonable and compliance can be demonstrated. It is a valid requirement for future transport aircraft providing what Lockheed believes is a cost effective means to protect the aircraft from wind loads when parked.

Recommendation

Accept the Requirement as stated.

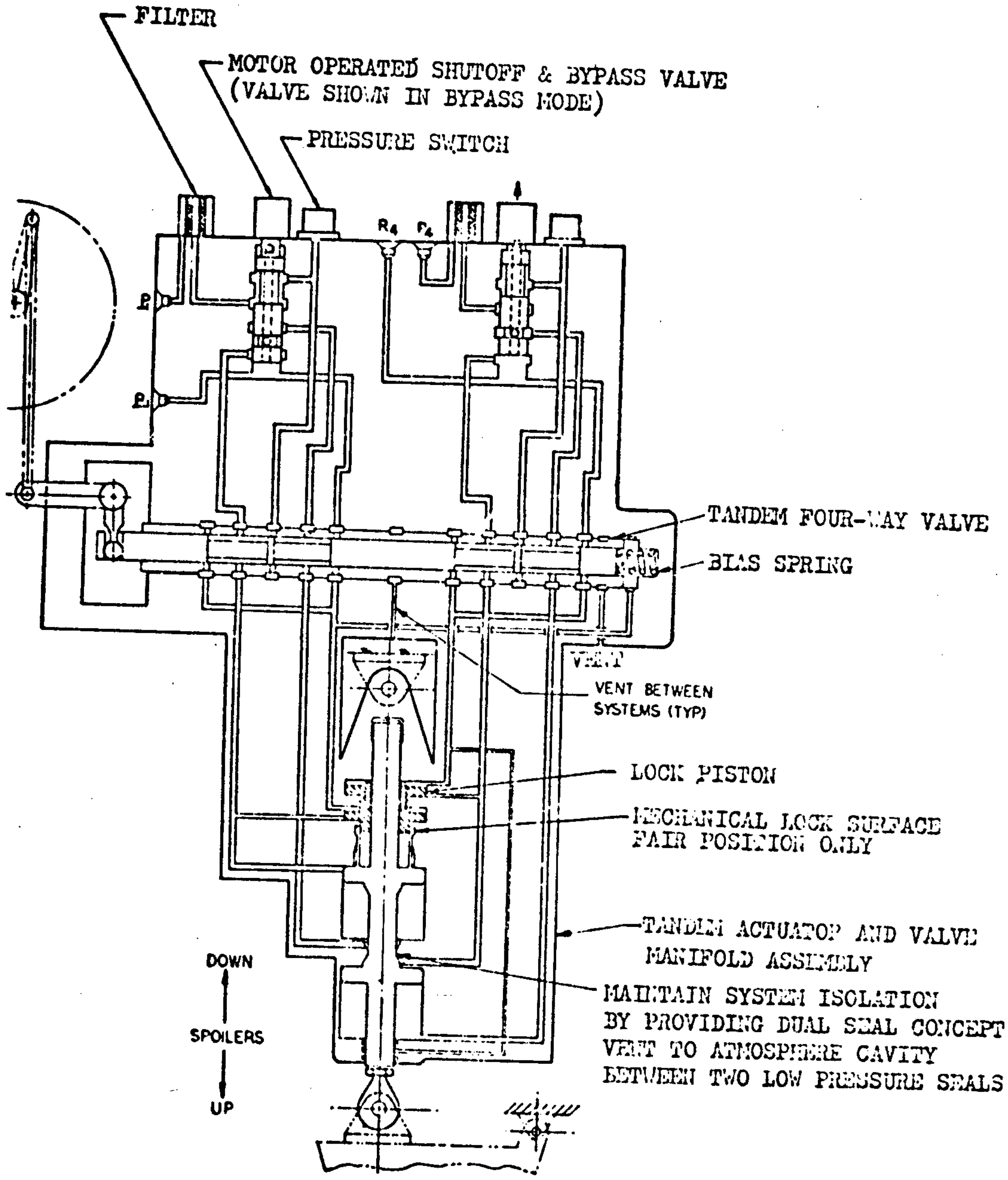


FIGURE 1 (3.2.6.7.2). HYDRAULIC SCHEMATIC GROUND SPOILER ACTUATOR

Requirement

3.2.6.7.2.1 Control Surface Locks. Where control surface locks are used, the lock system shall be internal within the airplane. External locks may be used for helicopter rotors. The locks shall either engage the surfaces directly or lock the controls as near to each surface as practicable and shall be spring loaded to the unlocked position. Control surface locks shall be designed to preclude attempting take-off with controls locked.

Comparison

The only surface mechanical locks used in the C-5 flight controls are contained in the ground spoiler actuators. These locks are engaged at all times except when ground spoilers are intended to be extended such as during a rejected take-off or a short field landing. Ground spoilers are not intended for use after lift-off; therefore, the locks are rightly engaged in flight. Hydraulic gust locks integral with the servo actuators are used when required at the locations.

Discussion

The requirement is valid and can be met economically.

The use of the C-5A ground spoiler actuator integral lock does, however, indicate a need to revise the specification and users guide to add requirements for those applications of the flight controls which are employed only prior to lift-off and/or after touch down and which normally should remain faired during flight.

Recommendation

Revise the last sentence of the requirement to add "except for those controls which are intended for use only on the ground."

Requirement

3.2.6.7.2.2 Protection Against Inflight Engagement of Control Surface Locks. Control surface ground gust locks and their controls shall be designed to preclude their becoming engaged during flight.

Comparison

Only C-5A ground spoilers employ mechanical surface locks integral with surface actuators.

Discussion

Care was taken in the C-5A design to assure that ground spoiler actuator locks remain engaged from lift off to touch down.

Recommendation

Restate the requirement as follows:

"3.2.6.7.2.2 Protection Against Inadvertent Engagement or Disengagement of Control Surface Locks. Ground gust locks for control surfaces and their controls shall be designed to preclude their becoming engaged during flight. Locks for control surfaces intended for use only during the take-off roll or landing roll and their controls shall be designed to preclude their becoming disengaged in flight."

Requirement

3.2.6.7.3 Control Surface Flutter and Buzz Prevention. All flight control surface actuation systems controlling surfaces which are not dynamically balanced shall be effectively irreversible or provided with sufficient damping to prevent flutter, buzz, or other relative dynamic instabilities for all operating modes and meet the requirements of MIL-A-8870. No active powered compensation technique or mechanization designed to artificially increase effective stiffness, damping, or natural frequency shall be used without prior approval of the procuring activity.

Comparison

C-5A control surfaces are statically balanced to varying percentages with the exception of the inboard elevator control surfaces. The inboard elevators are interconnected mechanically and each is provided with an unpowered mechanical-hydraulic damper assembly. The balancing and dampers preclude the possibility of any dynamic instability including flutter and buzz when operating the aircraft with inboard elevator control servo-actuators unpowered. No active powered compensation techniques are used to artifically increase effective stiffness, damping or natural frequency.

The C-5A FCS design meets the requirements of MIL-A-8870 which required the aircraft to be free of any flutter, buzz, divergence, and other dynamic aeroelastic, aerothermo-elastic, and aeroservoelastic instabilities of the aircraft and its relevant components at all speeds up to 1.15 V_L for all design ranges of altitudes, thermal conditions, and maneuvers where losses in rigidity could occur. See the validation discussion for Paragraph 3.1.11.2 for additional substantiating data.

Discussion

This is a valid requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. The "Users Guide" (AFFDL-TR-74-116) discussion on this requirement is excellent and is a good example of the type of coverage which should be provided for user information. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.7 Component Design

3.2.7.1 Common Requirements

3.2.7.1.1 Standardization. Where practical, contractor designed equipment which has been approved for use in some models of aircraft shall also be used in later model airplanes if the installation and requirements are similar.

Tolerances shall be such that interchange of any LRU with any other part bearing the same part number shall not require resetting of parameters or readjustment of other components in order to maintain overall tolerances and performance.

Comparison

The C-5A FCS did not use any Lockheed designed equipment used on earlier model airplanes.

Discussion

The requirement is a worthy design goal which should reduce FCS costs for any particular airplane when applicable. The C-5A FCS and component designs were significantly different from previous Lockheed designs because of the larger size of the aircraft. This required the use of all new Lockheed systems concepts and mechanization. Many of the previously designed components were reviewed in efforts to standardize equipment. However, systems dynamic performance and loading requirements prevented their use. Where possible, unhanded FCS servo assemblies and common components within them were used to achieve desired standardization. Care was taken in the design of these LRU multi-use assemblies and components to assure their physical and functional interchangeability without adjusting or resetting them. The requirement is well stated and is applicable to large transport type aircraft.

Recommendation

Accept as stated.

Requirement

3.2.7.1.2 Interchangeability. Like assemblies, subassemblies, and replaceable parts shall meet the requirements of MIL-I-8500 regardless of manufacturer or supplier. Items which are not functionally interchangeable shall not be physically interchangeable unless specifically approved by the procuring activity.

Comparison

C-5A interchangeability and replaceability requirements for purchased components are contained in Lockheed's Specification D4M90000, General Engineering Requirements for Vendor Designed Equipment, Appendix III. Interchangeability and replaceability general requirements are contained in Section 3.3.7 of the Air Vehicle Specification Document CP 40002-1A. These C-5A requirements satisfy the intent of this paragraph of MIL-F-9490D.

Discussion

Adequate controls were established to assure parts manufacture in accordance with Lockheed engineering data, information and control media. Where like components were used in different assemblies, such as a servo valve in a manifold, adequate assembly functional tests were conducted to assure that performance of each assembly met its requirements. Where changes to interchangeable parts were required during the C-5A development, part numbers were controlled and changed to assure appropriate control of the final configurations.

Care was taken through the use of design and dimensional controls and other means to assure that only physically and functionally interchangeable parts could be interchanged. The requirement is clearly stated and its application to large transport type aircraft is reasonable.

Recommendation

Accept the requirement.

Requirement

3.2.7.1.3 Selection of Specifications and Standards. Specifications and standards for necessary commodities and services not specified herein shall be selected in accordance with MIL-STD-143.

Comparison

Individual Lockheed C-5A control component procurement specification documents and D4M90000, General Engineering Requirements for Vendor Designed Equipment specify the application of MIL-STD-143 in the selection of specifications and standards for necessary commodities and services not otherwise specified.

Discussion

The C-5A met the requirement which is reasonable.

Recommendation

Accept "as is."

Requirement

3.2.7.1.4 Identification of Product. Equipment components, assemblies, and parts of flight control systems shall be identified in accordance with MIL-STD-130.

Comparison

CEI Detail Specification, CP 40002-6B, requires that C-5A equipment identification and markings be in accordance with MIL-M-25047, MIL-STD-130 and AFSCM80-1.

Discussion

TheC-5A met or exceeded the stated requirement. The requirement is reasonable and applicable to future transport aircraft.

Recommendation

Accept "as is."

Requirement

3.2.7.1.5 Inspection Seals. Corrosion resistant metallic seals shall be provided at all strategic locations to indicate assembly inspection and any unauthorized disassembly.

Comparison

The C-5A FCS used low weight Alucast seals at the required locations. These seals are aluminum alloy and have been proven to be satisfactory by environmental testing and actual service.

Discussion

The requirement is reasonable and is met by the C-5A. It is subject to interpretation, however, since the term "corrosion resistant metallic seals" may be interpreted to mean "corrosion resistant steel seals" in which case Lockheed believes the requirement would be too restrictive. The "Users Guide" does not clarify it.

Recommendation

Revise the requirement as follows:

> "3.2.7.1.5 Inspection Seals. Suitable corrosion and wear resistant seals shall be provided at all strategic locations...."

Requirement

3.2.7.1.6 Moisture Pockets. All components shall avoid housing designs which result in pockets, wells, traps, and the like into which water, condensed moisture, or other liquids can drain or collect. If such designs are unavoidable, provisions for draining shall be incorporated.

Comparison

The C-5A conforms to this requirement. Water traps were generally avoided and drain provisions were incorporated where condensed moisture might accumulate.

Discussion

The requirement is reasonable and desirable.

Recommendation

Accept "as is."

Requirement

3.2.7.2 Mechanical Components. Mechanical components not covered by design requirements specified elsewhere within this specification shall be designed in accordance with applicable requirements in: Government and Industry specification, in the order of precedence specified in MIL-STD-143; in AFSC Design Handbooks DH 2-1, DN 3B1, Mechanical Flight Controls; and DH 1-2, General Design Factors; and the following:

Comparison

The C-5A flight control system (FCS) mechanical component design criteria not covered by design requirements specified by definitive functional requirements were otherwise designed in accordance with the detail component requirements of Contract End Item specifications CP 40002-1A, CP 40002-6B, and related detail specifications therein. The FCS mechanical systems such as the cable systems and push rod systems and their related components were designed to criteria which meet or exceed the design practices specified in the AFSC-DH handbook sections for mechanical flight controls and general design factors. The FCS components were designed to minimum weight and size and the system design used the minimum number of parts required for that function.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept the specification "as is."

Requirement

3.2.7.2.1 Bearings. Flight control system bearings shall be selected in accordance with AFSC Design Handbook DH 2-1, Chapter 6, Airframe Bearings, and the following.

3.2.7.2.1.1 Antifriction Bearings. Approved type ball bearings in accordance with MIL-B-6038, MIL-B-6039, and MIL-B-7949 shall be used throughout the flight control system, except as indicated in the following paragraphs. Bearing installation shall be arranged in such a manner that failure of the rollers or balls will not result in a complete separation of the control. Where direct axial application of control forces to a bearing cannot be avoided, a fail-safe feature shall be provided.

3.2.7.2.1.2 Spherical Bearings. Where space or other design limitations preclude the use of antifriction bearings, spherical-type, self-lubricating plain bearings in accordance with MIL-B-81820, or spherical or special-type all-metal bearings in accordance with MIL-B-8976 with adequate and accessible provisions for lubrication, may be used.

3.2.7.2.1.3 Sintered Bearings. Sintered type, or oil impregnated bearings shall not be used in those parts of the flight control systems which have slow moving or oscillating motions. Their use in fast moving rotating applications, such as in qualified motors and actuators, is permissible. Bearings shall conform to MIL-B-5687.

Comparison

The C-5A flight control system (FCS) mechanical and hydraulic actuation systems, in most applications, utilized MS type bearings which were vendor supplied and had been designed to the Military Specifications referenced above and which were specified in the component design criteria of the Contract End Item specification CP 40002-6B. Special, spherical bearings, such as the primary surface servo actuator bearings, which were subjected to high load and cyclic conditions, were generally designed by the component or sub-system vendor and had to utilize, and in some cases develop, an advance in the state-of-the-art to meet the demanding requirements. An example is the development of special bearing liner material for the spherical bearings used in the surface actuator rod ends. The design criteria of the C-5A bearing design was MIL-F-9490C which allowed the use of these special spherical bearings subject to the approval of the Air Force.

All of the FCS bearings had to be designed and tested to an endurance life cycle requirement and an environmental life requirement in accordance with MIL-STD-810. Strict maintainability requirements produced many bearings which required little or no lubrication. The C-5A bearing design criteria followed the design guidelines discussed in AFSC CH 2-1, Chapter 6.

Discussion

The requirements as stated are good and compliance can be easily demonstrated. The C-5A design complies with the intent of the requirement as it stands. However, the specifications as stated may be restrictive, depending on interpretation, in utilizing and advancing the state-of-the-art technology in bearing design. Another area of concern is the magnitude of "direct axial application of control forces" discussed in 3.2.7.2.1.1. There is almost always some axial load involved, but in many cases it is of low magnitude with respect to the bearing axial load capability and therefore not sufficiently significant to require the fail-safe design feature. The specification, as modified below, is recommended for future transport type aircraft.

Recommendation

In order to define a magnitude of axial force, modify Paragraph 3.2.7.2.1.1 last sentence to read:

> "Where a significant (50% of the bearing's axial load limit) direct axial load application of control forces to a bearing cannot be avoided, a fail-safe feature shall be provided."

In order to allow for improved state-of-the-art technology, add another paragraph as follows:

> "3.2.7.2.1.4 Special Bearing Applications. Where the design and/or operational criteria dictate that the contractor must use a bearing design which does not meet all the requirements of Paragraphs 3.2.7.2.1, 3.2.7.2.1.1., 3.2.7.2.1.2., or 3.2.7.2.1.3., the design is subject to the approval of the procuring agency."

Requirement

3.2.7.2.2 Controls and Knobs. Aircrew controls shall be shaped and located per the requirements of AFSC Design Handbook DH 2-2. Control knobs shall be designed and spaced per the requirements of AFSC Design Handbook DH 2-2 and MIL-K-25049.

Comparison

The aircrew controls arrangement is suggested in AFSC Design Handbook DH 2-2. The C-5A arrangement is shown in Figures 1(3.2.7.2.2) and 2(3.2.7.2.2).

MIL-STD-203 Military Standard Cockpit Controls Location and Actuation for Fixed Wing Aircraft and AFCS Design Handbook 2-2 were required for the C-5A design. The C-5A control knobs and levers were shaped and located to meet these requirements.

Discussion

The MIL-F-9490D requirement is reasonable and allows the designer the freedom he needs to locate controls in the space available.

Recommendation

Accept "as is".

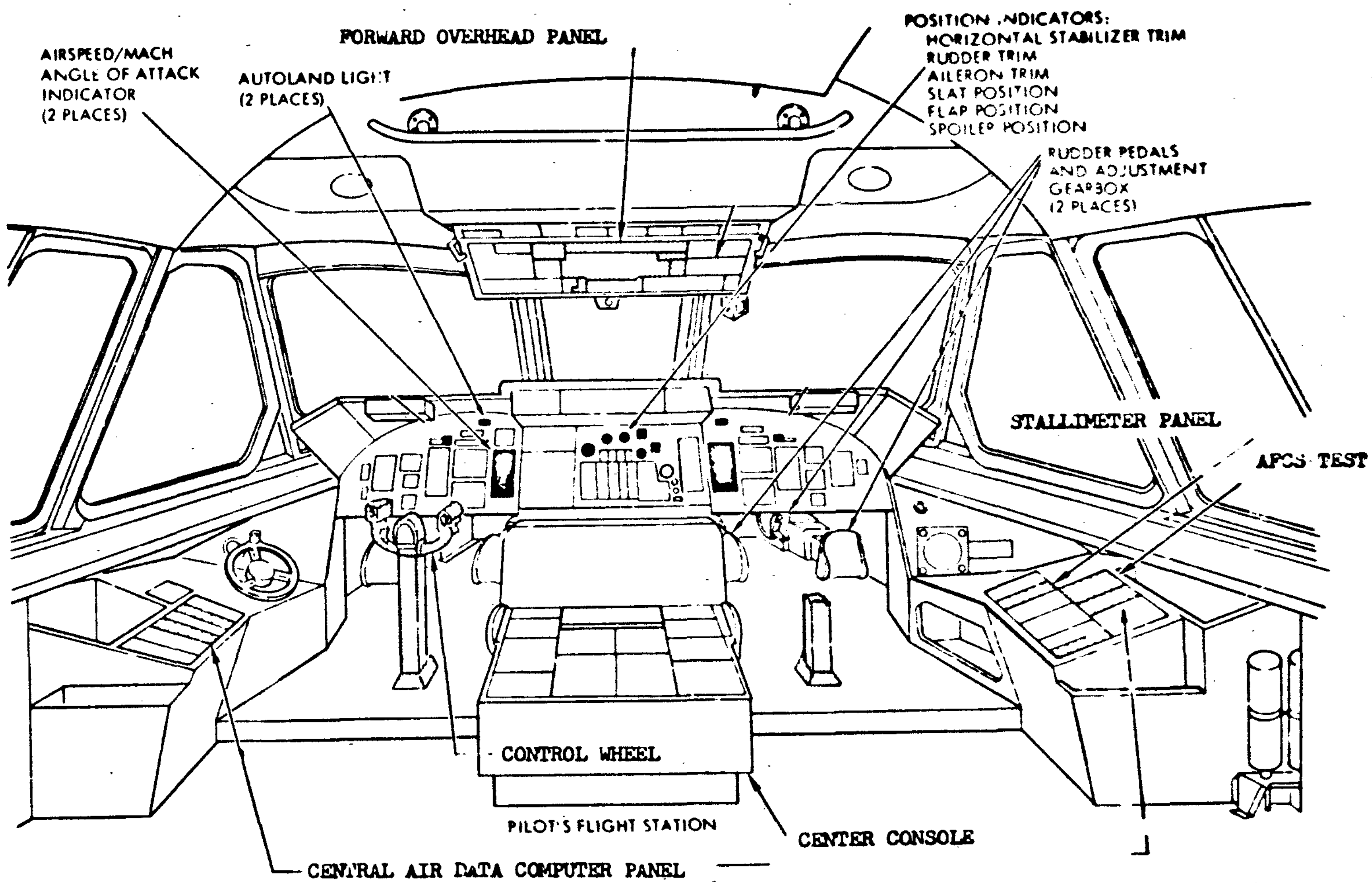


FIGURE 1 (3.2.7.2.2). C-5A FLIGHT STATION ARRANGEMENT

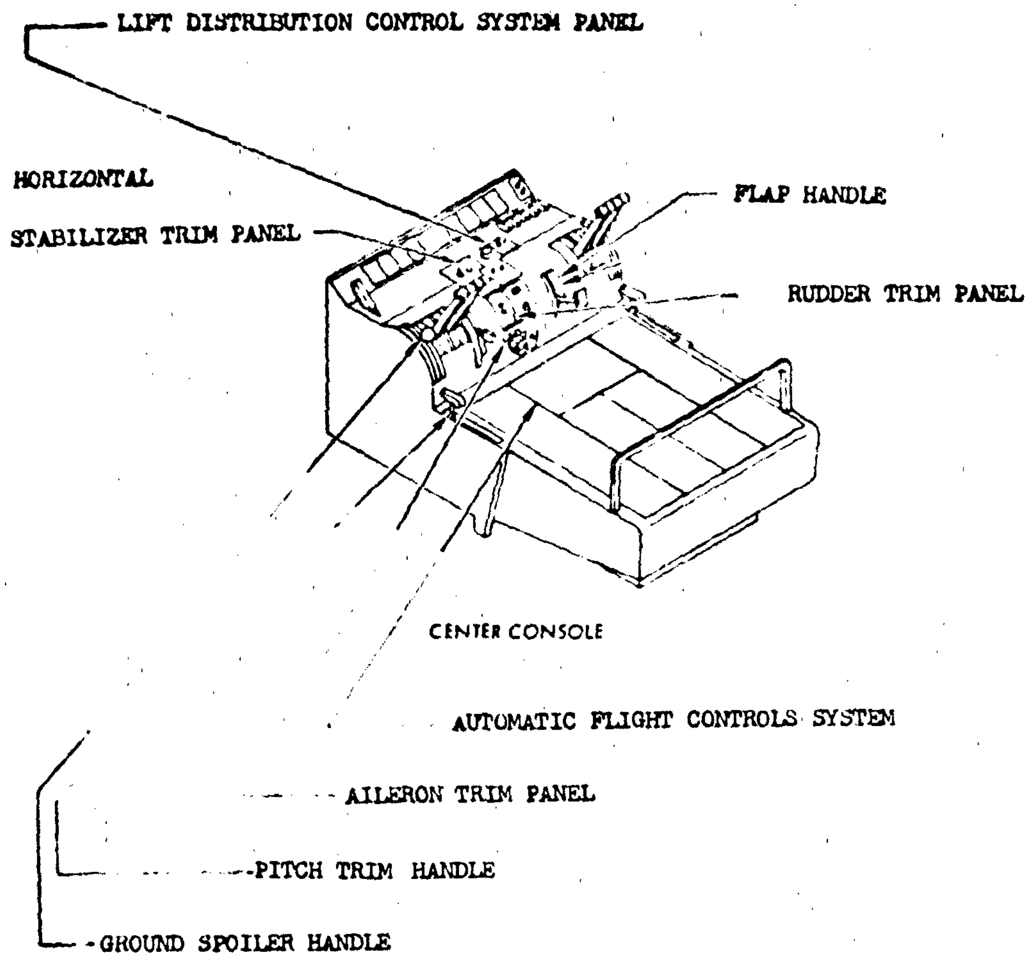


FIGURE 2 (3.2.7.2.2) C-5A FLIGHT STATION CENTER CONSOLE

Requirement

3.2.7.2.3 Dampers. Each damper shall be completely defined by a detail specification. Control stick dampers shall be designed so that they can be overpowered by the pilot in the event of failure or malfunction. Damping requirements for surface dampers shall be based upon the anticipated flutter frequency, but the endurance requirements shall be based upon the same criteria established for the surface control actuators. Detail design of hydraulic dampers shall conform to the applicable requirements of MIL-C-5503. All joints, connections, and bearings shall be designed to prevent the degree of wear which can cause unacceptable freeplay.

Comparison

The C-5A has an inboard elevator surface damper assembly which is defined by a specification document and an envelope drawing. The assembly was designed to cover a frequency range of 0-16 CPS which includes the expected flutter frequency of 2.8 CPS. The qualification test for the assembly incorporates an Endurance Life Test which includes the 5,000,000 cycle spectrum of the C-5A elevator system servo assembly and a total of 5,000,000 cycles at low deflection angle ($\pm$ 1°, $\pm$ 2° and $\pm$ 5°).

The damper assembly is designed to the applicable paragraphs of MIL-C-5503 and after the Endurance Life Test must still meet the specification damping requirements at the expected flutter frequency.

Discussion

The C-5A Inboard Elevator Control System and Surface Damper meets the requirements of this paragraph. The requirement is reasonable and can be demonstrated practically.

Recommendation

Accept "as is."

Requirement

3.2.7.2.4 Structural Fittings. All structural fittings used in flight control systems shall comply with the design requirements specified in AFSC Design Handbook DH 1-2, Design Note DN 4B1, Design Requirements, and where applicable, the design considerations specified in Design Note DN 4B2, Forgings and Castings.

Comparison

The C-5A flight control systems (FCS) structural fittings were designed to the guidelines and military specifications defined in Contract End Item specifications CP 40002-1A and CP 40002-6B. The C-5A design requirements and guidelines were more definitive than the requirements of AFSC Design Handbook DH 1-2, Design Notes, DN 4B1 and DN 4B2.

All materials used were in accordance with the requirements of MIL-Handbook-5.

Castings were designed to meet the requirements of MIL-C-6021, MIL-A-8860 and AFSCM 80-1. Most casting applications had a sample statically tested to ultimate load. No standard casting was used in any FCS primary load carrying member.

Forgings were designed to the requirements of MIL-F-7190, MIL-I-6868, MIL-A-22771, MIL-T-9047 and QQ-A-367.

Discussion

This is a good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Accept "as is."

Requirement

3.2.7.2.5 Lubrication. Where applicable, lubrication fittings in accordance with MIL-F-3541, MS15002-1 and -2, or NAS 516 shall be installed to provide for lubrication in accordance with MIL-STD-838. NAS 516 fittings are restricted to nonstressed areas only.

Comparison

The C-5A flight control system components which were not lubricated with hydraulic fluid were otherwise lubricated in accordance with MIL-STD-838. The lubrication fittings were in accordance with MIL-F-3541 and MS15001 and were readily accessible for servicing and replacement. Where possible permanently lubricated and liner type bearings were used to reduce the periodic servicing.

Discussion

This is a generally good requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft. It is recommended that "nonstressed area" in the last sentence is really intended to be "low stressed area."

Recommendation

Restate the last sentence of the requirement as follows:

"Nas 516 fittings are restricted to low stressed areas only."

Requirement

3.2.7.3 Electrical and Electronic Components. Electrical and electronic components not covered by design requirements specified elsewhere within this specification shall be designed in accordance with MIL-E-5400, MIL-E-7080, MIL-STD-454, MIL-STD-461, MIL-W-5088, MIL-M-7969, MIL-M-8609, and the following:

Comparison

This is a general requirement and cannot be directly compared to the C-5A. All these specifications were imposed on the C-5A AFCS where they were applicable. The C-5A meets the intent of this paragraph with the exception that solderless wrap wiring was authorized for internal wiring of computers and panels.

Discussion

Two of the specifications listed in this requirement, MIL-E-5400 and MIL-E-7080, are general specifications and call out the other specifications that are listed. MIL-E-5400 requires MIL-STD-454, MIL-STD-461, MIL-W-5088, MIL-M-7969, and MIL-M-8609 to be met. MIL-E-7080 requires MIL-W-5088, MIL-M-7969 and MIL-M-8609 to be met.

The C-5A AFCS used solderless wrap wiring for chassis wiring of the autopilot and SAS. This requirement as written does not authorize the use of solderless wire wrap and therefore is too strict.

This requirement, including all subparagraphs, could be interpreted as pertaining only to components not covered elsewhere in this specification. It is recommended that the subparagraph requirements should pertain to all FCS electrical and electronic equipment. The requirement as revised below is applicable to future transport aircraft with appropriate stringency and compliance is easily demonstable.

Recommendation

A. Revise the requirement as follows:

"3.2.7.3 Electrical and electronic components not covered by design requirements specified elsewhere within this specification shall be designed in accordance with MIL-E-5400, MIL-E-7080. All electrical and electronic components that are part of the FCS shall meet the following requirements (meaning requirements 3.2.7.3.1, 2, 3, 4, 5, 6, and 7)."

3. Add new requirement to this section as follows:

"3.2.7.3.7 Solderless Wrap Wiring. Solderless wrap wiring, for internal wiring of the computers/panels, shall conform to MIL-STD-1130 except that insulated 30 AWG solid copper wire may be used. The cables and wire shall be adequately supported and the insulation shall meet the temperature requirements as specified in MIL-W-5088. The insulation at the terminal posts shall have a cutting quality at the terminal of a minimum of 400 grms. tested at 500 volts."

Additional Data (For "Users' Guide")

The C-5A uses solderless wrap wiring for chassis wiring of the autopilot, SAS and ALDCS. No major problems have occurred due to the use of wirewrap. This type of construction should be considered for production on AFCS computers due to the fact that computer controlled wiring can be used for construction. Care should be exercised in the quality control, and sufficient qualification testing should be performed to insure compatibility with the proposed aircraft.

Requirement

3.2.7.3.1 Dielectric Strength. Leakage current shall not exceed 10 milliamps when a dielectric stress voltage of 1,200 volts, 60 Hz, is applied for 1 minute between insulated circuits and between circuits and case; and there shall be no insulation breakdown. When 500V DC is applied between isolated circuits and the case or connector shell for a period of 10 seconds, the resistance shall be at least 50 megohms. When a component or connector has a lower design voltage limitation, the test shall be run at an appropriate lower voltage as defined by the component specification.

Comparison

Electrical and electronic components of the C-5A FCS were subjected to insulation dielectric tests related to their usage.

The AFCS components were subjected to the following test:

Insulation Dielectric

The device as assembled for air vehicle use; i.e., all cards and covers installed, shall have 500V DC applied between each external pin and the chassis for pins not having a circuit to the chassis, and between circuits isolated from each other and the chassis. The insulation resistance in each case shall not be less than 100 megohms.

The flight director computer was subjected to the dielectric tests of TSO-C52a which included overpotential and insulation resistance tests. TSO-C52a requirements are as follows:

6.1 Dielectric. Each instrument shall be tested by the methods of inspection listed in Paragraph 6.1.1 and 6.1.2.

6.1.1 Insulation Resistance. The insulation resistance measured at 200 volts DC for five seconds between all electrical circuits connected together and the metallic case shall not be less than 5 megohms. Insulation resistance measurements shall not be made to circuits where the potential will appear across elements such as windings, resistors, capacitors, etc., since this measurement is intended only to determine adequacy of insulation.

6.1.2 Overpotential Tests. Equipment shall not be damaged by the application of a test potential between electrical circuits, and between electrical circuits and the metallic case. The test potential shall be a sinusoidal voltage of a commercial frequency with the RMS value of five times the maximum circuit voltage or per Paragraphs 6.1.2.1 or 6.1.2.2 whichever applies. The potential shall start from zero and be increased at a uniform rate to its test value. It shall be maintained at this value for five seconds, and then reduced at a uniform rate to zero.

Since these tests are intended to assure proper electrical insulation of the circuit components in question, these tests shall not be applied to circuits where the potential will appear across elements such as windings, resistors, capacitors, etc.

6.1.2.1 Hermetically sealed instruments shall be tested at 200 volts RMS.

6.1.2.2 Circuits that operate at potentials below 15 volts are not to be subjected to overpotential tests.

The electrical components installed on the C-5A hydraulic servo-actuators were subjected to a different dielectric test than were the electronic components. This was done because of their construction and location. The C-5A hydraulic servo-actuator electrical component test requirements are as follows:

Insulation Dielectric Test. The unit shall be subjected to a high-potential insulation test to determine compliance. The test voltage shall be 1,500 volts, rms, for 1 minute or 1,800 volts, rms, for a period of 1 second at commercial frequencies, except that the low power circuits, 50 volts or less, shall be subjected to a test voltage of 1,000 volts, rms, for a period of 1 minute or 1,250 volts, rms, for a period of 1 second. Capacitors and electrical devices likely to be damaged by application of these potentials may be disconnected. However, prior to assembly, a test voltage of twice the maximum peak voltage encountered during normal operating conditions or a minimum of 100 volts, whichever is greater, shall be applied for a period of 1 minute. There shall be no evidence of arc or insulation breakdown. These potentials shall be reduced by 25 percent for a second or succeeding test on the same unit.

As a result of meeting the above requirement, it is believed that the C-5A electrical components installed on hydraulic servo-actuators can meet this MIL-F-9490D requirement. Electronic components will meet the 500V DC requirement, but it is felt that they would not meet the 1200 VAC requirement.

Discussion

This requirement is not applicable to all electrical and electronic FCS components on existing and future aircraft. This requirement should take into account the components usage, construction, and location. For example, a computer located in the pressurized area need not be constructed to withstand 1200 VAC on its terminal as required by this paragraph. This would impose an unrealistic requirement and increase complexity of future equipment. The requirement to use a 60 Hz test voltage is too restrictive since some of the components are purchased outside of the United States where 60 Hz

commercial voltages may not exist and would require unnecessary purchase of special test equipment to replace available commercial frequency equipment.

This requirement should be revised to be a general requirement that covers future aircraft as well as taking into account differences between electrical and electronic flight control components. This new requirement should take into consideration the dielectric strength requirements of MIL-F-9490C, FAA TSO's, and this requirement. The MIL-F-9490C requirement is given below for reference and the TSO requirement is given in the comparison paragraph of this evaluation.

<u>MIL-F-9490C Requirement:</u>

4.1.3.6.2 <u>Dielectric Strength</u>. Each circuit of electrical and electronic components shall be subjected to a test equivalent to the application of a root mean square test voltage of three times the maximum (but not less than 500V) surge dc, or maximum surge peak ac, voltage to which the circuit will be subjected under service conditions. The test voltage shall be of commercial frequency and shall be applied between ungrounded terminals and ground, and between terminals insulated from each other, for a period of 1 minute. Tests shall be accomplished at normal ground barometric pressure. No breakdown in insulation or air gap shall occur. Circuits containing capacitors or other similar electronic parts which may be subject to damage by application of the above voltages shall be subjected to twice the surge peak (but no less than 100V) operating voltage for the specified period. If the maximum peak operating voltage is greater than 700V, the rms value of the test voltage shall be 1050V greater than 1.5 times the maximum peak operating voltage. Electrical and electronic components shall also be tested for resistance to air gap breakdown at the maximum altitude specified in the altitude test.

<u>Recommendation</u>

Revise the requirement as follows:

"3.2.7.3.1 <u>Dielectric</u>. The insulation selected and used to isolate all current-carrying element from each other and the case of the unit shall be such as to withstand all operating voltages encountered in service while installed in the air vehicle. The requirement of Paragraphs 3.2.7.3.1.1 and 3.2.7.3.1.2 shall be met."

Add:

"3.2.7.3.1.1 <u>Insulation Resistance</u>. Each circuit of electrical and electronic components shall be subjected to test voltage of not less than 500 VDC. Voltage is to be applied between isolated circuits and the case or connector shell for a period of 10 seconds. The insulation

resistance shall not be less than 50 megohms and there shall be no evidence of breakdown in insulation. Insulation resistance measurements shall not be made to circuits where the potential will appear across elements such as windings, resistors, capacitors, etc. Where easily removable elements are involved, such as plug-in modules, they shall be removed prior to the individual circuit test.

"3.2.7.3.1.2 Overpotential Tests. Electrical and electronic components shall be subjected to a test potential between isolated circuits and between circuits and the case or connector shell. The test potential shall be a sinusoidal voltage of a commercial frequency with a root mean square (rms) value of five times the maximum surge peak voltage. The voltage shall be applied five seconds for electronic components and one minute for electrical components and then reduced at a uniform rate to zero. In the case of electrical components, the minimum test voltage shall be 1000 volts rms. During the test the leakage current shall not exceed 10 milliamps and no damage shall occur. Hermetically sealed units shall be tested at 200 volts rms and circuits that operate at potentials below 15 volts are not to be subjected to overpotential tests. This test shall not be applied to circuits where the potential will appear across elements such as windings, resistors, capacitors, etc. Where easily removable elements are involved, such as plug-in modules, they shall be removed prior to the individual circuit test."

Requirement

3.2.7.3.2 Microelectronics. When used, microelectronic devices shall conform to the provisions of MIL-M-38510.

Comparison

Microelectronic devices and circuits were used in the C-5A AFCS. The Lockheed procurement specification stated, "Microelectronic and integrated circuits shall be used throughout the construction of the equipment where reliability and maintainability characteristics are enhanced and system performance characteristics are not degraded. The use of microelectronics and integrated circuits shall be limited to multiple usage items and low-cost, throw-away units and shall provide justification for such recommendations made." The C-5A vendors were required to meet the system or LRU performance and qualification requirements for all parts installed. It is unknown whether all the microcircuits used were designed to meet MIL-M-38510.

Discussion

This is a good requirement and the C-5A would probably meet the intent of MIL-M-38510. A rewording of this requirement is required since the specification, MIL-M-38510, covers microcircuits and does not cover microelectronic devices in a general manner. This could be misunderstood to allow only the use of microcircuits per MIL-M-38510.

Recommendation

Revise the requirement as follows:

> "3.2.7.3.2 Microelectronics. When used, microcircuits used in microelectronic devices shall conform to the provisions of MIL-M-38510."

Requirement

3.2.7.3.3 Burn-In. All electronic LRU's shall receive a minimum of 50 hours burn-in operation and testing prior to assembly, or after assembly if such is more meaningful, prior to installation. Performance after burn-in shall be within specified tolerances.

Comparison

The C-5A originally required the AFCS vendor to perform to Reliability Production Acceptance Requirements which required that the first 20 systems be subjected to 70 hours each of burn-in testing for a total of 1,400 hours. This testing was to include functional cycling with concurrent vibration and temperature cycling. Because of development changes encountered during the reliability demonstration testing, the C-5A AFCS requirement was changed to 168 hours of burn-in testing on each LRU.

The C-5A meets this requirement.

Discussion

This requirement can be misinterpreted to include preliminary acceptance testing as part of the 50 hour minimum burn-in. From experience, we have found that the first production units will have the most failures and take the most time to pass the acceptance test procedure. This time will be reduced as changes are made and the acceptance test procedure modified. Even if automatic acceptance testing is to be used, which is usually not perfected until the first production run is completed, the acceptance test time on the first complicated LRU's could exceed the basic 50 hours.

The last sentence of this requirement leaves a question as to what the specified tolerances are. All LRU's that are subjected to burn-in testing should meet the normal acceptance test procedure.

The C-5A will meet this requirement with the recommended changes.

Recommendation

Revise the requirement as follows:

> "3.2.7.3.3 Burn-In. All electronic LRU's shall receive a minimum of 50 hours burn-in testing prior to installation and after original acceptance testing. Performance after burn-in shall meet the normal ATP requirements."

Additional Data (For "Users' Guide")

The sentence indicated by the left vertical sideline should be added to the background information and "Users' Guide."

Requirement

3.2.7.3.4 Switches. The design of special electric/mechanical switches, other than toggle switches, shall be subject to the approval of the procuring activity.

Comparison

The C-5A uses push button as well as toggle and rotary switches. All these are state-of-the-art designs and are not considered to be special electric/mechanical switches.

The C-5A meets this requirement.

Discussion

This requirement is considered to be too lenient by excluding special toggle switches from being approved. There are many types of toggle switches available today which are capable of being used in the FCS. It is felt that if a special design of a toggle switch is required then that design should be approved by the procuring activity.

This requirement as modified is appropriate for future aircraft.

Recommendation

Revise the requirement as follows:

> "3.2.7.3.4 Special Switches. The design of special electric/mechanical switches shall be subject to the approval of the procuring activity."

Requirement

3.2.7.3.5 Thermal Design of Electrical and Electronic Equipment. Wherever feasible, components shall be designed with heat-dissipating efficiency adequate to allow simple conductive, radiation, and free convection cooling utilizing the ambient heat sink to maintain the components within their permissible operating temperature limits. Operation under specified conditions shall not result in damage or impairment of component performance.

Comparison

The C-5A electrical and electronic LRU's were designed for free convection cooling. For example, due to the location of a control panel in a stagnant area, a fan was added to the unit to reduce the temperature of the push button switches on the front panel.

Discussion

This requirement is considered appropriate and well stated and is appropriate for future aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.7.3.6 Potentiometers. Resistive variable voltage dividers shall not be used in dynamic motion applications such as sensor outputs or feedback output devices without specific approval by the procuring agency.

Comparison

The C-5A utilizes Linear Variable Differential Transformed (LVDT) in the flight control power units for stability augmentation feedback. The autopilot servos use synchros for position feedback. The C-5A meets the intent of this paragraph by not using potentiometers in AFCS dynamic motion applications.

Discussion

This is a good requirement and is applicable for future transport aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8 Component Fabrication. The selection and treatment of materials processing and assembly may be in accordance with established contractor techniques in lieu of the following requirements, upon approval by the procuring activity.

3.2.8.1 Materials. When Government specifications exist for the type material being used, the materials shall conform to these specifications. Nonspecification materials may be used if it is shown that they are more suitable for the purpose than specification materials. These materials shall have no adverse effect upon the health of personnel when used for their intended purposes. This requirement shall be met for all probable failure modes and in the required environments.

3.2.8.1.1 Metals. Metals used in flight control system components shall be selected in accordance with the criteria and requirements specified in AFSC Design Handbook DH 1-2, Design Note DN 7A1, Metals.

3.2.8.1.2 Nonmetallic Materials. Nonmetallic materials, shall conform to the requirements specified in AFSC Design Handbook DH 1-2, Design Note DN 7A1, Nonmetals.

Comparison

3.2.8 Component Fabrication. The C-5A FCS producibility (fabrication) considerations, as established by CEI CP 40002-1A, were an integral part of the design development of the air vehicle. The functional systems production breakdown provided for the maximum efficiency for the established production rates. The basic design minimized the requirements for new or high skill levels, special training, complex preventive and corrective maintenance techniques, special ground support (AGE) equipment, and special tools. Value Engineering guidelines in accordance with MIL-V-38352 and approved Lockheed manufacturing process specifications were adhered to.

The FCS design considered the capability of personnel and equipment available for procurement and fabrication that permitted a high quality of workmanship. All fabrication was subject to inspection by the cognizant company and government designated inspection activities. All fabrication was controlled by the contractor's design specifications, drawings, process specifications, standards, and quality assurance controls to meet the objectives of the air vehicle design.

3.2.8.1 Materials.
3.2.8.1.1 Metals.
3.2.8.1.2 Nonmetallic Materials. The C-5A FCS used standard, proven and economical parts, materials, and processes to the maximum extent; consistent with the reliability, maintainability, and performance requirements of the FCS specifications.

Raw materials were procured on the basis of military specifications (as a

first preference) and approved commercial and company material and process specifications used in an order of selection which follows MIL-STD-143. Restricted use of special materials and products was determined by cost effective trade studies which considered the functional requirements, weight and detail design requirements. The selection of metals used the guidelines and mechanical properties as defined in MIL-Handbook-5 except for certain approved non-military specifications. Standard metals were used where practical and any special metals used were selected by cost effective trade studies which included considerations of cost, weight, delivery schedule, reliability, structural integrity and endurance life. Special approval was required for use of steels heat treated above 220 ksi and the use of new "Super Alloys."

Nonmetallic materials used in the FCS design included elastomeric materials, plastics, fiberglass, ceramics, lubricants, etc., selected from approved material sources. In addition to the primary design requirements the nonmetallic materials had to show maximum resistance to ozone, fluid environment (i.e., hydraulic fluid) and general natural and induced environments, i.e., temperature, humidity, weathering, etc.

Discussion

The requirement for metal selection is not definitive enough and should include the guidelines of MIL-Handbook-5. Otherwise, this is a valid requirement which has been satisfied by the C-5A FCS design, can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Revise Paragraph 3.2.8.1.1, "Metals," as follows:

> "Metals used in flight control system components shall be selected in accordance with the criteria and requirements specified in MIL-Handbook-5 and AFSC Design Handbook DH1-2, Design Note DN7A1, Metals."

Requirement

3.2.8.1.3 Electric wire and cable. Electrical wire cables containing up to seven conductors shall be constructed in accordance with MIL-C-27500. Airframe wire bundles may be constructed in accordance with contractor developed techniques provided such construction is approved by the procuring activity.

Comparison

The C-5A FCS electrical wiring and cable installations were designed to conform to the requirements of MIL-W-5088 and CEI specification CP40002-1A and CP40002-6B. For applications below 600V for electrical subsystems and avionics hookup wiring used the guide lines of MIL-C-27500 with the applicable wire and cable specifications such as MIL-W-81044, MIL-W-22759, MIL-C-7078, etc. Applications were determined by the temperatures of the wiring environment. In special high temperature application specifications were created by Lockheed and approved by the Air Force.

Discussion

The C-5A FCS wiring meets this specification in that MIL-C-27500 is a general wire cable fabrication specification. For applications of more than seven conductors deviations to the basic requirement must be granted by the procuring agency. The C-5A wiring application had authority to use more than seven conductors if required.

This requirement can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.2 **Processes**

3.2.8.2.1 Construction Processes. Heat treating, adhesive bondings, welding, brazing, soldering, plating, drilling, and grinding of high strength steels, materials inspection, castings, forgings, sandwich assemblies, and stress corrosion factors used in the fabrication of flight control system components shall comply with the requirements specified in AFSC Design Handbook DH 1-2, Design Note DN7B1, Construction.

Comparison

The FCS components were fabricated using approved, proven, and controlled processes which were clearly specified on all applicable drawings and fabrication instructions. All applicable construction processes were considered as an integral function of FCS component design and fabrication as noted in the validation discussions for Paragraphs 3.2.8, "Component Fabrication," and 3.2.8.1, "Materials." Where applicable all processes were in accordance with a military specification in addition to the company process specifications. Examples of the applicable military specifications were: MIL-H-6875 for heat treating steel, MIL-H-60 8 for heat treating aluminum, MIL-STD-454 for soldering, MIL-S-5002 plating, MIL-C-6021 for castings, MIL-F-7190, MIL-I-6868 for forgings, and MIL-A-9067, MIL-A-25463, MIL-A-5090 for adhesive bonding.

Discussion

The specification can be interpreted to imply that the applicable construction processes are limited to those listed. The specification should allow the use of any applicable contractor approved process. This is a valid requirement which has been satisfied by the C-5A FCS design, can be demonstrated, and should be specified for all future transport type aircraft.

Recommendation

Revise the requirement as follows:

> Add the following to clarify the requirement: "Applicable construction processes may include contractor approved and controlled processes already in use which satisfy the intent of the requirement.

Requirement

3.2.8.2.2 Corrosion Protection. All flight control system component parts, except those inherently resistant to corrosion in the operational environments, shall be finished per AFSC Design Handbook DH 1-2, Design Note DN7B2, Corrosion.

Comparison

The C-5A FCS elements and components were suitably finished so as to provide protection from corrosion. The finishes and coatings used complied with the requirements of MIL-F-7179, Type II, in addition to other more specific application requirements. Metal surfaces were treated to the requirements of MIL-S-5002 and provide adequate corrosion resistance.

Aluminum surfaces were anodize coated in accordance with MIL-A-8625. Steel (noncorrosion resistant) components were cadmium-titanium (anodic) plated. Cadmium plating was in accordance with QQ-P-416.

The use of dissimilar metals (as defined in MS33586) in direct contact, was prohibited. If dissimilar metals were required to be joined, their faying surfaces were adequately insulated to assure protection from electrolytic corrosion. Sealing of dissimilar metals fay surfaces was accomplished with MIL-S-8802 sealant or equivalent.

All permanently installed fasteners penetrating areas subject to corrosive conditions were wet installed with an applicable corrosion inhibitor. Drainage holes were provided where practical to prevent entrapment of water.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design; it can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.2.3 Fabrication of electrical and electronic components. The applicable requirements in AFSC Design Handbook DH 1-6, Design Note DN3H1, Electrical/Electronic Safety Design Considerations, relating to the fabrication of electrical and electronic components shall be met.

Comparison

The C-5A FCS electrical and electronic components materials, parts, and processes were designed to the fabrication requirements of the CEI specifications contained in CP40002 and applicable military specifications to the extent specified therein. The applicable military specifications included MIL-E-5400 for electronic equipment, MIL-E-25499 for electrical systems, and MIL-E-7080 for electrical equipment for aircraft.

Design considerations included those referenced in AFSC Design Handbook DH1-6, Design Note DN3H1 covering such areas as material selection, fabrication, maintenance, environmental considerations, etc. Included were detail considerations such a lightening protection, electrical bonding, grounding, connections, terminal blocks, receptacles, etc. Applicable construction processes such as referenced in the validation discussion for paragraph 3.2.8.2.1 were used.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and should be specified for all future transport type aircraft. However, this requirement is listed under the heading of paragraph 3.2.8.2 'Processes' and since the subject deals with 'Fabrication of electrical and electronic components', it should be included under the heading of paragraph 3.2.8.3.3 'Assembly of Electronic Components'.

Recommendation

Renumber the requirement as follows:

from 3.2.8.2.3

to 3.2.8.3.3.6

Requirement

3.2.8.3 Assembling

3.2.8.3.1 Mechanical Joining. Individual parts may be mechanically joined with removable fasteners, or by riveted or threaded connections, or by qualified methods for permanent joining.

Comparison

The C-5A FCS elements and components used approved mechanical joining techniques which included removable fasteners, riveting and threaded connections. Mechanical joining design was in accordance with CEI specification CP 40002-1A using the guidelines of AFSCM80-1 and Lockheed standard components and design practices. Joining techniques were selected based on cost effective trade studies considering the FCS weight and performance goals. Emphasis was placed on using standard and conventional methods of joining structural applications with primary considerations of stress distribution, fatigue, and stress corrosion. Fasteners used on systems that may have affected safety of flight, i.e., FCS, engine controls, etc., and all FCS elements using rotating joints incorporated a close tolerance bolt and a dual locking device consisting of at least a self-locking cotter pinned castellated nut. Critical FCS joints used a self-retaining bolt plus a self-locking cotter pinned castellated nut. Examples include the input and feedback rods on the primary servo actuators. Where practical, fasteners with a proven service life were used. All single bolt connections in the FCS used bolts which were one-fourth inch diameter or larger. All removable fasteners are positively retained; i.e., all nuts are at least self-locking.

All FCS elements with adjustments have the adjusting featu mechanically retained; i.e., jam nuts may have a locking tab and/or are lock-wired. Threaded joints are positively locked to prevent load reversal at the threads and have adequate wrenching and holding provisions. Minimum thread engagements are marked by an inspection hole. Joining by rivets is used on permanently attached components.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.1.1 Joining with Removable Fasteners. All removable fasteners shall be selected and used in accordance with the applicable requirements specified in AFSC Design Handbook DH 1-2, Design Notes 4A1, General Requirements, 4A3, Bolts, Nuts, and Washers; 4A4, Screws; 4A5, Pins; and 4A6, Other Fasteners except as follows:

a. Bolts smaller than 1/4 inch in diameter shall not be used to make single-bolt connections or connections essential to proper functioning of the component.

b. Each removable bolt, screw, nut, pin, or other removable fastener, the loss of which would degrade operation below FCS Operational State III, shall incorporate two separate locking or retention devices either of which must be capable of preventing loss of the fastener by itself and retain it in its proper installation with the other locking or retention device missing, failed, or malfunctioning. Where self-retaining bolts are used, their selection and installation shall be within the limitations of MS33602, and only one type shall be used in any given system.

c. No self-locking nut may be used on any bolt subject to rotation in operation unless a nonfriction locking device is used in addition to self-locking device.

d. Lockbolts listed in AFSC Handbook DH 1-2, Design Note 4A5, Swaged-Collar-Headed Straight Pins and Collars, may be used for fastening applications not requiring removal on the aircraft.

Comparison

The C-5A FCS removable fasteners were selected and used in accordance with the system functional requirements, standard industry practices, AFSCM80-1 handbook practices and the general mechanical joining practices noted in the validation discussion for Paragraph 3.2.8.3.1. The fasteners were chosen to resist vibration and acoustic forces. The fasteners used were predominantly military specification standard parts with emphasis on minimizing the number of sizes and types. The fasteners incorporated corrosion protection practices, such as surface finishes in accordance with MIL-F-7179, minimum use of dissimilar metals in accordance with MIL-STD-889, Cadmium plating per QQ-P-416, etc.

Fasteners were never designed to be subjected to the fully rated tensile and shear load simultaneously. Threaded fasteners were designed to never impose any appreciable bearing loads on the threads.

Screw thread inserts were never used in a primary load application. Self-tapping screw inserts were not used. Fasteners were designed to the allowable load margins in MIL-Handbook-5.

Close tolerance fasteners were used in applications subject to high shear

Requirement

loads, load reversals, and to reduce system free play. Where possible the design practice of installing bolt heads up, forward and inboard was followed to reduce the probability of falling out when the retaining nut was not in place. Removable fasteners were located to allow easy access and removal for maintenance where possible. Self-locking, castellated nuts are used on all bolts which are used as a rotating joint and which may affect safety of flight, as noted in validation discussion for Paragraph 3.2.8.3.1. Castellated nuts use cotter pins per MS24665. Safety wiring of fasteners was in accordance with MS33540. Threaded fasteners that carry a structural load and that are used to join two or more components shall be the bolt-thru type secured at the threaded end by a safety nut.

On any pivot fastener, the threaded portion was not subjected to any appreciable bearing load and was designed to prevent loss of the locking feature from rotation.

All single bolt connections were one-fourth inch diameter or larger.

Plain washers were used in most removable bolt applications to protect the surface from injury and to reduce the stress of the joint by increasing the bearing area.

Fastener threads in high strength steel parts were in accordance with MIL-S-8879; most other threads were in accordance with MIL-S-7742.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design; it can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.1.2 Joining with Rivets. Rivets for all riveted joints shall be selected and used in accordance with the requirements specified in AFSC Design Handbook DH 1-2, Design Note 4A2, Rivets.

Comparison

The use of rivets for FCS components was limited and generally applied to permanently attached components. Special rivets were used as shear joints on certain FCS interconnecting rods as shown in Figure 3 (3.2.3.2.5). All rivet joints were in accordance with the allowable loads specified in MIL-Handbook-5 for the material used. Rivets were selected and used in accordance with the applicable standard practices defined in the validation discussions for Paragraphs 3.2.8.3.1 and 3.2.8.3.1.1.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design; it can be demonstrated, and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.1.3 Threaded Joints. All threaded joints shall be provided with adequate wrenching and holding provisions for assembly and disassembly of the joint before and after service use. Internal screw threads and external rolled threads shall be in accordance with the thread form requirements of MIL-S-8879. Pipe threads shall not be used.

Comparison

The FCS threaded joints have adjustment features that are mechanically retained; i.e., jam nuts have a locking tab and/or are lockwired. The threaded joints are positively locked to prevent load reversal at the threads and have adequate wrenching and holding provisions for assembly and disassembly of the joint before and after service use.

Figure No. 3 (3.2.3.2.5) shows a typical rod end with adjustment and locking features. Screw thread forms were in accordance with MIL-S-8879 requirements. Pipe threads were not used in the FCS component design.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design; it can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.2 Joint Retention. All adjoining parts shall be secured in a manner that will preclude loosening when subjected to internal or external loads or vibration.

3.2.8.3.2.1 Retention of Threaded Joints. All threaded joints which carry critical loads shall be positively locked in the assembled position so that load reversal at the threads is prevented. The use of jam locknuts alone is not a positive locking means unless lockwired or otherwise restrained.

3.2.8.3.2.2 Retention of Removable Fasteners. Unless restrained from moving by the attachment of adjoining parts, all removable fasteners shall be positively locked in place. Self-locking externally threaded fasteners shall not be used except within the limitations specified in MS15981, and self-locking nuts shall not be used except within the limitations specified in MS33588. All other types shall incorporate positive locking means or be safetied with cotter pins in accordance with MS24665, where temperature and strength permit, or be safety wired. Cotter pins and safety wiring shall be installed in accordance with MS33540.

Comparison

All the C-5A FCS component assemblies and adjoining elements are retained in a manner which precludes loosening when subjected to internal or external loads or vibrations. Threaded joints which carry critical loads are positively locked to preclude load reversal at the threads. Figure No. 3 (3.2.3.2.5) shows a typical actuator rod end locking device. A locking washer is mechanically "keyed" to the actuator rod and rod end to prohibit rotation at the threads. In applications where jam locknuts are used as a locking device the jam nuts are safety wired together.

The retention of removable fasteners is discussed in the validation comparison for Paragraphs 3.2.8.3.1 and 3.2.8.3.1.1. All removable fasteners are positively locked in place; i.e., bolts are retained by nuts which are at least self-locking and in some applications a self-locking cotter pinned castellated nut is used. Cotter pins are used in accordance with MS24665. Safety wiring was accomplished in accordance with MS33540.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design, can be demonstrated, and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.2.3 Use of Retainer Rings. Retainer rings shall not be used to retain loaded parts unless the rings are positively confined by a means other than depending on internal pressure or external loads. They shall not allow freeplay which could result in structurally destructive action or fatigue failure of the retained parts or failure of gaskets or packings. Where used, retainer rings shall be commercially available types which can be installed and removed with standard tools.

Comparison

The C-5A FCS restricts the use of snap rings as follows.

Snap rings were not used in any application where improper installation or dislocation of the ring would cause a malfunction or failure of the unit. Snap rings were not used where the accumulation of tolerances would allow destructive end play or looseness. Snap rings were installed and removed with standard pin-type tools. Spirally wound locking rings were not used.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design, can be demonstrated, and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.3 Assembly of Electronic Components

3.2.8.3.3.1 Electrical and electronic part mounting. Electronic parts shall be mounted so that ease of producibility and maintainability is assured. Whenever feasible, parts such as resistors, capacitors, etc. shall be mounted in an even, regular, row-type arrangement. These parts shall be mounted on a base so that the leads do not cross other leads or connections. Heavy electronic parts and assemblies shall be solidly mounted so that adverse effects when subjected to vibration and shock are minimized.

Comparison

The C-5A FCS electrical and electronic parts were mounted in accordance with the installation requirements specified in CEI CP40002-6B. The applicable installation specifications included MIL-STD-704, MIL-E-7080, MIL-E-25499, MIL-I-8700 in addition to other CEI and systems specifications. The electrical and electronic systems used in the primary flight control systems, except for the power source, were designed with the objective of having no interconnection with any other electrical system. All electrical and electronic parts were mounted to provide easy access and maintainability. The mounting of small components such as resistors, capacitors, or terminals were arranged in even, regular rows and arranged on a base such that the interconnecting leads did not cross over other leads or terminal connections.

All equipment was mounted to minimize the effects of vibration, acceleration, shock and all other adverse effects of induced and natural environments.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design. The requirement can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.3.2 Shielding and bonding on finished surfaces. Nonconductive oxides or other nonconductive finishes shall be removed from the actual contact area of all surfaces required to act as a path for electric current and from local areas to provide continuity of electrical shielding or bonding. All mating surfaces shall be clean and shall be carefully fitted, as necessary, to minimize radio frequency impedance at joints, seams, and mating surfaces. The resultant exposed areas, after assembly at such joints or spots, shall be kept to a minimum.

Comparison

The C-5A electrical shielding and bonding was accomplished by methods defined by Lockheed process bulletins. The process required that all non-conducting coatings, films, and finishes be removed from the faying surfaces for all bonding straps, jumpers, ground wire terminations and bonds between two metal parts. The cleaned area did not permit the introduction of any corrosive agent and it provided a path for electric current continuity for the required shielding or bonding. The resultant assemblies exposed areas were minimized and chemically conversion coated for subsequent refinishing. All mating surfaces were configured to minimize radio frequency impedances.

Discussion

This is a valid requirement which has been satisfied by the C-5A design. The design can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.3.3 Isolation of redundant circuits. Redundant circuits shall be isolated from each other to preclude failure of one portion of the circuit from affecting any other circuit.

Comparison

The C-5A FCS electrical signal transmission was designed to the criteria goals of MIL-W-5088 and as defined in CEI specification CP40002, which in general were more stringent than the military specifications. The intent of these requirements was to guard against the susceptibility to single or multiple failures in the transmission signal paths. The FCS circuit wiring was designed to the design goal that redundant circuits be isolated physically and electrically such that a single failure would not cause the loss of both systems. Wires and cables were routed for maximum reliability and minimum interference and coupling between systems. Cables and junctions were identified in accordance with standard techniques and utilized dissimilar or "keyed" connections to assure separation and prevent cross connections of adjacent connections.

Discussion

The C-5A FCS wiring complies with the intent of this requirement as a design goal where it could be practically achieved on the redundant circuit of the SAS system, such that at least partial compliance was adhered to. However, the critical nature of this system was considered from the standpoint of failure modes.

The requirement can be demonstrated and should be applicable to all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.3.4 Electrical Connector Installation. The number of electrical connectors shall be kept to a minimum within the required limitations for separation of redundant circuits. Connectors shall be mounted to preclude nuisance warning indications and intermittent operation when subjected to applicable temperature differentials, vibration, and shock. They shall be polarized so that it is impossible to mismate them on a particular piece of equipment.

Comparison

The C-5A FCS electrical connector selection and installation followed the basic requirements dictated by specifications such as MIL-E-5400 and MIL-E-7080 as noted in the validation discussions for Paragraphs 3.2.7.3, 3.2.8.3.3.1 and 3.2.9.4 for electrical and electronic components and installations. The number of electrical connectors were kept to a minimum within the requirements for isolation of redundant circuits as noted in the validation discussion for Paragraph 3.2.8.3.3.3 and the requirements for harness installation, component fabrication, and maintainability. All connectors were selected and mounted to comply with the applicable natural and induced environment criteria. All connectors were "keyed" to preclude the possibility of mismatch or cross connection during installation.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design. The requirement can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.8.3.3.5 Cleaning of Electrical Assemblies. All electrical assemblies shall be thoroughly cleaned of loose, spattered, or excess solder, metal chips, or other foreign material after assembly. Burrs, sharp edges and resin flash shall be removed.

Comparison

The C-5A FCS components were fabricated using approved processes and fabrication techniques as noted in the validation discussions for Paragraphs 3.2.8.2.1 "Construction Processes" and 3.2.8.2.3 "Fabrication of Electrical and Electronic Components." Fabrication, process, and quality assurance specifications required all electrical assemblies to be thoroughly cleaned of excess solder, metal chips, and foreign material after assembly. Burrs, sharp edges, and resin flash was removed as part of these fabrication techniques.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design. The requirement can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.9 Component Installation

3.2.9.1 Basic Requirements. Flight control system components shall be installed in compliance with the applicable requirements specified in AFSC Design Handbook DH 1-6, Section 3J, Flight Control Systems, including Design Note 3JX, Safety Design Check List, and as specified herein.

Comparison

The C-5A FCS component installations were designed to meet CEI CP 40002-1A and -6B and all related specification requirements therein. The bulk of the installation requirements contained in Paragraphs 3.2.1 "Pilot Controls and Displays" and 3.2.6 "Actuation" were met as noted in their validation discussions. In addition, FCS design practices the same as those required by AFSC Design Handbook DH 1-6, Section 3J, were followed as noted below. Design guideline philosophies included the following:

- o The preclusion of FCS installations problems and hazards related to adjacent systems and maintenance errors.
- o Design to minimize installation errors which could have resulted in equipment being incorrectly installed or failures resulting therein may have caused problems in adjacent as well as the primary systems.
- o Routing, separation, and installation of FCS components as noted in the validation discussion for Paragraph 3.2.9.2 "Locating Components."
- o Redundancy and failure criteria design approaches and levels determined on an individual basis within the flight envelope as defined in the validation discussions for Paragraphs 1.2.2, 1.2.3, and 3.1.3.1.
- o The FCS components were designed to comply with the functional requirements within the specified natural and induced environments; i.e., protection by surface treatment, component location, etc.
- o The FCS design achieved the functional characteristics required to assure meeting the reliability and maintainability requirements.

Other design considerations included:

- o In addition to system redundancy, the FCS was provided with jam override capability.
- o The FCS design precluded cross connection of control elements.

- The FCS design incorporated fouling prevention features; i.e., cable tension regulators for slack take-up, element guides, etc.

- Proper selection of FCS components was assured on the basis of their functional requirements.

- Redundant power control systems were provided such as the multiple hydraulic systems to power the hydraulic servoactuators.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design, can be demonstrated, and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.9.2 Locating Components. System components shall be located to provide direct routing of the control system signal and power transmission elements (cables, rods, lines, wires, etc.) in accordance with Design Note 3J1, Routing and Separation, only to the extent that the components and transmission elements are not exposed to undue hazards.

Comparison

In addition to the installation design guidelines outlined in the validation discussion for Paragraph 3.2.9.1, the following design guidelines were also used. FCS element routing was determined to assure adequate unrestrained motion envelopes, proper separation, most direct path, and minimal effect from the induced environmental effects from adjacent systems.

Separation of redundant and adjacent systems was provided to insure safety and survivability. Different hydraulic systems, powered by different engines, was provided for redundant hydraulic servoactuators. The normal and emergency power systems have different power sources; i.e., engine driven hydraulic pumps are backed up by a ram air turbine (RAT) driven hydraulic pump.

Other component installation considerations included mounting provisions to orient their position. Adequate spacing between FCS elements and adjacent structure or equipment was provided to insure that structural failure or buckling did not create binding or a hazard. The designs provided protection from jamming or fouling from dropped or loose articles.

Suitable drainage was provided for unwanted fluid accumulation.

Maximum separation was provided between fluid (hydraulic) lines, electrical wiring and equipment, oxygen lines, etc. Adequate access was provided for maintenance.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design, can be demonstrated in non-quantitative terms, and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.9.3 Installation in Fuel System Areas. All component installations in fuel system areas shall preclude the generation of sparks both during normal operations and possible abnormal and failure conditions.

Comparison

There are no FCS components located in a fuel system area which could generate sparks. The closest FCS installation adjacent to a fuel system area is the mechanically actuated leading edge slat screw jacks which are mounted from brackets on the wing box front beam.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design, can be demonstrated, and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.9.4 Electrical and Electronic Component Installation. In addition to the requirements specified in AFSC Design Handbook DH 1-6, Section 3J, the applicable requirements in Design Notes DN 3H1, Electrical/Electronic Safety Design Considerations and DN 3H2, Installation Safety Objectives, shall be met.

Comparison

The C-5A electrical and electronic flight control systems were installed in accordance with the applicable portions of MIL-STD-704, MIL-E-7080, MIL-E-25499, MIL-I-8700 and many more military specifications. Throughout the C-5A FCS specification, CP 40002-6B, there are many specific requirements that can be equated to this paragraph of MIL-F-9490D. The C-5A meets the intent of this requirement.

Discussion

The requirement is reasonable, desirable and applicable to future aircraft.

Recommendation

Retain the requirement as stated.

Requirement

3.2.9.5 Electrical and Electronic Equipment Cooling. If cooling augmentation is required, the installation of flight control electrical and electronic equipment colling shall be integrated with the cooling provisions for other electrical and electronic equipment. The requirements specified in AFSC Design Handbook DH 1-6, DN 3H1, Temperature shall be met.

Comparison

Electrical and electronic flight control equipment on the C-5A were designed for free convection cooling. Only the Automatic Flight Control Panel required a fan to be added to reduce the panel temperature. The C-5A meets the intent of this requirement.

Discussion

This requirement is reasonable, desirable, and applicable to future aircraft.

Recommendation

Retain the requirement as stated.

Requirements - Not applicable

3.3 Rotary Wing Performance and Design

3.3.1 Special MFCS Performance Requirements

3.3.2 Special AFCS Performance

3.3.2.1 Attitude Hold (Pitch, Roll and Yaw)

3.3.2.2 Heading Hold and Heading Select

3.3.2.3 Altitude Hold

3.3.2.3.1 Barometric Altitude Stabilization

3.3.2.3.2 Stabilization of Altitude Above the Terrain

3.3.2.4 Hover Hold

3.3.2.5 Vernier Control for Hovering

3.3.2.6 Groundspeed Hold

3.3.3 Special Design Requirements

3.3.3.1 MFCS Design

3.3.3.1.1 Control Feedback

3.3.3.1.2 Feel Augmentation

3.3.3.2 AFCS Design

3.3.3.3 Swashplate Power Actuators

3.3.3.3.1 Redundancy

3.3.3.3.2 Jamming

3.3.3.3.3 Frequency Response

3.3.3.4 Actuation Stiffness

3.3.3.5 Fatigue Life Design

3.3.3.5.1 Fail-safe

3.3.3.5.2 Display

3.3.3.6 Built-in-Test

Requirement

4. QUALITY ASSURANCE

4.1 General Requirements

4.1.1 Methods for Demonstration of Compliance. Flight Control system compliance with each of the applicable design requirements of this specification or the FCS specification defined by 4.4.2 shall be verified using one or more of the following methods. Except where a specific method is required, selection of the method of proof shall be made by the contractor subject to concurrence of the procuring activity.

4.1.1.1 Analysis. Compliance with requirements in cases where testing or inspection would be hazardous or otherwise impractical may be verified through analyses. These analyses may be linear or nonlinear and may include piloted and nonpiloted simulations, as defined by the FCS development plan.

4.1.1.2 Inspection. Compliance with requirements associated with referenced component specifications, the physical arrangement of parts, or the physical relationship of parts shall be verified by inspection of documentation or inspection of the physical installation. Documentation may include documents showing the qualification status of components which have been qualified to the requirements specifications, or drawings showing clearances or other physical relationships. The FCS development plan shall define those requirements to be verified through inspection. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection requirements as specified herein. Except as otherwise specified in the contract or order, the supplier may use his own or any other facilities suitable for the performance of the inspection requirements specified herein. The Government reserves the right to perform any of the inspections set forth in the specification where such inspections are deemed necessary to assure supplies and services conform to prescribed requirements.

4.1.1.3 Test. To the maximum extent feasible, compliance with the quantitative requirements of the FCS specification shall be demonstrated by tests. Tests shall include the laboratory, airplane ground and flight tests defined in the FCS development plan.

Comparison

For the C-5A aircraft, the FCS specification defined in Paragraph 4.4.2 is the CEI Detail Specification CP 40002-6B. Paragraph 4.1.3 of CP 40002-6B specifies that formal qualification demonstration be accomplished in accordance with FAA TSO-C9C and MIL-F-9490 as applicable. The subparagraphs which follow in the CEI specification specifically list which requirements in the design specification shall be qualified by inspection, analyses, demonstration and testing. The specification further delineates the exact procedures to be followed in testing compliance.

Discussion

The intent of this set of requirements is to allow the contractor to demonstrate compliance of the FCS with the specifications of this document in an optimum way as chosen by the contractor. Testing is the preferred method for demonstrating compliance, but it is not always optimum or even practical from cost and safety aspects. The inspection method offers the best means of compliance demonstration in cases of subsystems previously qualified to a detail specification and in cases of physical layout or characteristics which require visual observation. Analysis also has its place in compliance demonstration and it is employed effectively to reduce costs of showing compliance. In areas which would be too expensive or hazardous to demonstrate by testing, analysis is clearly the choice. Examples of areas where analysis is used extensively and effectively include structural load paths for flight controls components, reliability and failure analyses, turbulence response studies, and stability analysis. The requirements provide the contractor with the option of choosing which method or combination of methods is most suitable for evaluating compliance of his specific design with the requirements.

These requirements, which are applicable to the C-5A aircraft, are applicable to the design of future transport aircraft. The paragraphs are appropriate for their purpose as discussed above and need no changes.

Recommendation

Accept the requirement "as is."

Requirement

4.2 Analysis requirements. Where compliance with specification requirements through analytical predictions is used, the contractor shall define the major assumptions and approximations used and verify that the modeling and analysis procedures used are conservative. Verification shall normally require prior use and validation through comparison with flight, wind turnel or ground testing data. In all cases the contractor shall establish tolerances on analytical predictions used to demonstrate compliance with specification requirements. These tolerances shall reflect anticipated variations in system or component characteristics such as:

a. Parameters that change with temperature, atmospheric pressure and other environmental factors.

b. Parameters that change with failures or manufacturing tolerances.

c. Parameters that critically affect system performance or stability.

d. Parameters that are not accurately known (if they are significant).

e. Parameters that change as a result of aging or wear.

Comparison

The C-5A FCS CEI specifications required quality assurance confirmation of specification requirements through analysis techniques in addition to inspection, test, and demonstration as noted in the validation discussion of paragraph 4.1.1 'Methods for Demonstration of Compliance'. The requirements to demonstrate compliance by analysis was established considering reduced costs, impracticality, and areas otherwise too hazardous to show compliance. In some areas of compliance verification one or more of the other methods mentioned above were used in conjunction with the analysis. Examples of areas where analysis was used to show specification compliance included -

- FCS structural load paths and stress analysis.
- Maintainability and maintenance repair cycles.
- Primary, secondary, and automatic (including SAS) FCS specification compliance.
- Reliability
- Flight, ground, nuclear, and personnel safety.
- Failure effect analysis including power failure.
- Human performance.
- Value engineering.

The analysis tool was used extensively to derive parameters or data which were used in design areas which may have been used to establish basic criteria. Examples of these applications would include turbulence response studies and servo actuator stability analysis. Other examples are the C-5A flight simulation studies used during the design and development of the FCS which utilized math modelled parameters derived from analytical data and procedures.

All the FCS subsystem and component level analyses were required to verify compliance with such parameters as hydraulic servo actuator characteristics such as control valve pressure and flow metering, seals, and special materials properties and processing.

The basic analysis parameters utilized data which were known from previous airframe industry history and previous analysis or were derived from flight test, simulation, wind tunnel test, etc. The major assumptions and approximations were defined. Every reasonable effort was made to keep the modelling and analysis procedures conservative. The analysis tolerances were established to reflect anticipated variations in the system parameters.

Analysis parameters reflected residual influences. For example, the vibration analysis considered the environmental and operational effects on the aircraft such as altitude, speed, temperature, atmospheric pressure, etc. Other analysis parameter considerations were system failures, wear, performance, etc.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design. The requirement can be demonstrated in non-quantitative terms and should be specified for all future transport aircraft.

Recommendation

Retain the requirement as stated.

Requirement

4.2.1 Piloted Simulations. Piloted simulations shall be performed during FCS development. As a minimum, the following simulations shall be accomplished:

a. Piloted simulations using computer simulation of the FCS prior to hardware availability.

b. Piloted simulations using actual FCS hardware prior to first flight.

Comparison

Development of the C-5A flight control systems included computerized simulations of the systems prior to hardware availability. The systems math modelled were longitudinal and lateral-directional stability augmentation, autopilot, flight director, automatic go-around, auto-throttle, and the all weather landing system. Piloted simulations using actual FCS hardware were also performed on the C-5A program. The stability augmentation systems hardware was evaluated this way prior to first flight. However, the autoland system, for instance, was involved in piloted simulation on the C-5A Ironbird after first flight but prior to its evaluation inflight.

Discussion

Piloted six-degrees-of-freedom flight simulation studies during development and design of the flight control systems are essential. In most cases, piloted simulation using math modelled FCS will have an important input into the FCS design and should therefore be conducted prior to the final hardware design fix date. The importance of scheduling a development program this way is becoming more apparent with the advent of CCV where the impact on handling qualities in addition to system performance and aircraft safety is increasingly important.

Requiring piloted simulations prior to first flight using actual hardware will be less important in some cases depending on the flight test plans and schedules. It is essential that the hardware be tested for design compliance and pilot familiarity on an Ironbird Simulator, but unless the system is flight critical, or the flight test program is very short, there is no need and in fact it may be a cost panelty to tie hardware in-the-loop simulation to the first flight date. Instead, flight simulation with the hardware should be dovetailed into the flight test program. Flight critical items such as SAS and CCV should indeed be tied into the Ironbird Simulator and checked out including pilot-in-the-loop testing prior to first flight. A FCS such as an autoland system is not usually scheduled to be flight tested until well after first flight; therefore, major subsystems of that system may not be in completed hardware form at the first flight date.

Recommendation

Modify 4.2.1.a to read as follows:

> "Piloted simulations using computer simulation of the FCS prior to the hardware design freeze."

Modify 4.2.1.b to read as follows:

> "Piloted simulations using actual FCS hardware prior to the time each FCS subsystem is scheduled for flight evaluation."

Requirement

4.3 Test Requirements

4.3.1 General Test Requirements

4.3.1.1 Test Witness. Before conducting a required test, the contractor shall notify an authorized procurement activity representative. An orientation briefing on specific test goals and procedures shall be given procuring activity observers prior to any required test sequence to be monitored by an observer.

Comparison

A C-5A contractural document 3-17-1 concerning the Category I Test Plan stipulates that the procuring agency may witness any or all component qualification tests at their discretion, or they may delegate this function to the AFPRO in accordance with the program schedule. Prior to conducting a C-5A system or component qualification test or test sequence on Engineering Critical Items at Lockheed-Georgia, notification was given the designated observer in the local AFPRO. In addition, the Vendors who supplied flight control system equipment to Lockheed were required to notify the Defense Contract Administration Services (DCAS) representative prior to initiating any qualification test or test sequence. In summary, Lockheed is in compliance with this requirement.

Discussion

The requirement is considered somewhat vague on two points. First, it is not clear what is meant by the phrase "required test." Secondly, there is no clear indication who would define "required tests" or when such a requirement would be given. It is recommended that the paragraph be revised to clarify these two points.

Recommendation

Revise Paragraph 4.3.1.1 as follows:

> 4.3.1.1 Test Witness. Qualification tests on flight control systems and components as well as other required tests shall be contractually defined and included as a line item in the Contract Data Requirements List (CDRL). To the extent required by line item in the CDRL, the contractor shall notify the procuring agency, or its designated representative, prior to conducting contracturally required tests. In the event the procuring agency or its designated representative elects to monitor a test or test sequence, an orientation briefing on specific test goals and procedures shall be given the designated observer prior to the test.

Requirement

4.3.1.2 Acceptance Tests. Appropriate FCS acceptance tests will be defined by the procurement detailed specification.

Comparison

The component specification documents covering C-5A vendor-designed flight control equipment, including Engineering Critical Components, require that the preparation of Acceptance Test Procedures (ATP's) be in accordance with Section 18 of D4M90000 and that the procedures include certain minimum performance test requirements as defined in the component specification documents. D4M90000 is a Lockheed-Georgia document that defines the general engineering requirements for vendor-designed equipment. Lockheed complies with the intent of this requirement.

Discussion

Lockheed establishes at least the minimum acceptance test requirements (ATR's) in the detailed or component specification to insure component suitability and compliance with the overall system performance and operational requirements. Lockheed either prepares the acceptance test procedure (ATP) or approves the ATP when the component design and test responsibility is subcontracted to a vendor. There may be other tests which a vendor includes in his ATP which, although not required by the ATR, provide the vendor with an extra measure of assurance in certain areas. In any case, Lockheed as the prime contractor always is responsible for approving the vendor's ATP.

Recommendation

Retain the requirement as stated.

Requirement

4.3.1.3 Instrumentation. Accuracy of instruments and test equipment, used to control or monitor test parameters shall have been verified since its last use prior to initiation of the sequence of design verification tests. All instruments and test equipment used in conducting design verification tests shall:

a. Conform to laboratory standards whose calibration is traceable to the prime standards at the U.S. Bureau of Standards.

b. Be accurate to within one third the tolerance for the variable to be measured.

c. Be suitable for measuring the test parameter(s).

d. Be verified no less frequently than every 12 months.

Comparison

The C-5A FCS test and monitoring instrumentation specifications were imposed thru the Contract End Item requirements primarily in Document 3-17 for the C-5A Category I Test Plan.

Document 3-17 outlines the C-5A subsystem instrumentation and data acquisition requirements and methods.

All instrumentation was maintained and periodically certified with calibration which was traceable to the National Bureau of Standards in accordance with MIL-C-45662, MIL-Q-9858 and in consonance with other applicable Lockheed standards. Calibration of all instrumentation was performed to the best of current industry and state-of-the-art practices with the maximum accuracy commensurate with available standards and the requirements established by the test need or intended usage.

Document 3-17 contained listings of all the basic test parameters to be measured and the applicable accuracy and tolerances, in addition to a listing of all the test equipment required to supply this data.

Confirmation calibration of the instrumentation was periodically checked and was always verified prior to initiation of a sequence of design verification tests.

The accuracy of the instrumentation was a minimum of one fourth the tolerance for the variable to be measured and in some applications this ratio was tighter because of the critical nature of that particular test variable.

The frequency of calibration verification depended on the use of the equipment and how stable it was. Some equipment was verified monthly and other equipment only once in five years.

Discussion

This C-5A instrumentation met and in most cases exceeded this specification requirement with the possible exception of the instrumentation verification frequency.

There are military specification standards available which could and should be used to standardize certain instrumentation calibration and laboratory practices. The actual tolerances for instrumentation and frequency of calib ration verification should be determined by the required use of the equipment and the stability of the test equipment and should be mutually agreed upon between the procuring activity and the contractor. This requirement should be applicable to all future transport type aircraft but should be revised to require application of appropriate military specifications and to delete tolerance and frequency requirements which must be keyed to the application.

Recommendation

Revise the requirement as follows:

Revise 'a'. to read -

a. Conform to laboratory standards in accordance with MIL-C-45662 and MIL-Q-9858 except as noted below, and whose calibration is traceable to the prime standards at the U.S. Bureau of Standards.

Revise 'b'. to read -

b. The accuracy tolerances for the variables to be measured shall be submitted for the approval of the procuring agency.

Revise 'd'. to read -

d. Verification frequency shall be established based on the test tolerances and sensitivity and stability of the test equipment and shall be subject to approval of the procuring agency.

Requirement

4.3.1.4 Test Conditions. The contractor shall establish operation test conditions which accurately represent system in-service usage throughout the applicable flight phases and flight envelopes defined in accordance with MIL-F-8785 or MIL-F-83300.

Comparison

The C-5A FCS was subjected to operational test conditions which were representative of the in-service usage throughout the applicable mission requirements of the Contract End Specifications CP40002-1A and -6B. The test conditions for each component reflected aircraft usage throughout the natural environment of all conditions of weather and climate in any area of the world using MIL-STD-210, FAA-T30-C9C, and MIL-F-9490, as applicable.

Environmental areas included solar radiation, lightning, fog, ice-fog, dew, hail, icing, sleet, frost, salt spray, sand, clouds, and fungus. The FCS was subjected to test conditions simulating the effects of the induced environments resulting from the operational envelope of the aircraft operating in the natural environments mentioned above. Induced environments considered such areas as high temperature, temperature shock, vibration, mechanical shock, noise, acceleration, explosive or corrosive vapors, nuclear radiation, sound, EMI limits, and exhaust gases.

The mission requirements imposed applicable ground operating and non-operating conditions, flight phases and flight envelopes derived from MIL-F-8785 requirements to the extent defined in the CEI specifications. The methods of imposing the applicable test conditions are discussed to some extent in the validation of Paragraphs 4.3.2.1 "Component Tests" and 4.4.1 "Flight Control System Development Plan."

The FCS was subjected to testing under the influence of many of the environmental conditions during the aircraft flight environmental testing programs. In addition, all the FCS components were subjected to environmental qualification tests which generally used the test methods and procedures defined in MIL-STD-810. In some cases, more specific or detailed tests were defined in the detail component specification.

Discussion

This is a valid requirement insofar as it goes and it has been satisfied by the C-5A FCS design. The requirement can be demonstrated and should be specified for all future transport type aircraft with a revision to cover ground operating and non-operating conditions.

Recommendation

Revise the requirement as follows:

"4.3.1.4 Test Conditions. The contractor shall establish operation

test conditions which accurately represent system in-service usage throughout the applicable ground conditions, flight phases and flight envelopes defined in accordance with MIL-F-8785 or MIL-F-83300."

Requirement

4.3.2.1 Component Tests. All components shall be qualified to the applicable component specification by individual tests, by proof of similarity to qualified components which are qualified under conditions applicable to the specified operating conditions, by testing in system design verification tests, or suitable combinations of these methods. Component qualification requirements shall be based upon their use in the specific vehicle and its associated environment. Environmental test methods and procedures shall be selected from MIL-STD-461 or MIL-STD-810. The contractor shall generate additional methods and procedures where MIL-STD-461 or MIL-STD-810 are inadequate for the planned aircraft usage. Wear life 3.1.12 shall be demonstrated at the component level except where system wear life is more meaningful due to component interaction.

Comparison

All the C-5A FCS components were qualified to the quality assurance requirements of CEI specification CP 40002-1A and -6B. Each FCS component was qualified to the requirements of the applicable detail component specification document. Formal quality assurance qualification verified compliance with each design requirement in the component specification by inspection, analysis, demonstration, and/or component testing as noted in the validation discussion for Paragraph 4.1.1.

The majority of the design requirements were verified by completion of a series of individual tests which were applicable to the specified component operation criteria under the specified test conditions. Formal qualification acceptance was based on successful completion of the component test requirements. However, most of the FCS components were subjected to additional functional and endurance testing for total system design verification such as the "iron bird" FCS simulation tests. Additional specialized reliability testing was conducted on certain FCS components. Inspection always constituted one phase of the qualification procedure. Inspection of the component assembly, subassembly, and related parts confirmed conformity to the requirements of the applicable drawings and specification documents. Inspection of the components was conducted before and after each major phase of the qualification testing. In addition to visual inspection, the component acceptance test as described in the validation discussion for Paragraph 4.3.1.2 was used as an inspection tool. Inspection of the endurance life cycled test specimen included a complete teardown and inspection before and after completion of the test.

Other design requirements which required analysis were, in some cases, verified by demonstration and/or testing. For example certain maintainability requirements were first shown by analysis to meet the maintenance repair cycle and time guarantees and subsequently verified by performance of the actual maintenance functions on the components.

Particular component test requirements, as noted, were based on the specific aircraft component design criteria and the associated environmental test conditions as specified in MIL-STD-810. The electromagnetic interference

design test requirements were in accordance with MIL-STD-826 which has since been replaced by MIL-STD-461.

Component tests were divided into groups which included component performance characteristics, endurance life cycles, and environmental test requirements. Using a hydraulic servoactuator as the example component test specimen, the following types of tests were applicable:

> The component performance tests demonstrated compliance with design requirements for such areas as primary and secondary performance characteristics (actuator rates, operational check, limit load, hysteresis, friction, closed loop response, leakage, hydraulic manifold component tests, etc.)
>
> The test specimen was subjected to an endurance life cycle test to demonstrate the useful life of the component in meeting the requirements and compliance as noted in the validation discussions for Paragraphs 3.1.11.3 "Durability," 3.1.12 "Wear Life" and 3.2.6.4.3 "Actuating Cylinders."
>
> The test specimen demonstrated compliance with environmental requirements as noted in the validation discussion of Paragraphs 3.1.9.1 and 3.1.9.3 concerning the invulnerability to natural and induced environments. Environmental testing was conducted in accordance with the test methods of MIL-STD-810 and the detail component specification. Specific environmental tests included high temperature, low temperature, atmospheric, humidity, salt fog, sand and dust, fungus, explosive atmospheric, vibration, acceleration, shock, etc.

In some cases these test requirements were expanded upon to cover the particular type of equipment for peculiar environmental influences. For example, dielectric strength tests conducted were based on particular applications as noted in the validation discussion for Paragraph 3.2.7.3.1.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS design, can be demonstrated, and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

4.3.2.2 Functional Mockup and Simulator Tests. Where one of the first airplanes in a new series of aircraft will not be available for extensive testing of the FCS prior to flight of that model, an operational mockup which functionally, statically and dynamically duplicates the flight control system shall be constructed. For essential and flight phase essential flight controls, an accurate electrical representation shall also be provided. Production configuration components shall be used for all flight control system parts, and the hydraulic system shall be compatible with MIL-H-5440 test requirements. Primary aircraft structure need not be duplicated; however, production configuration mounting brackets shall be used and shall be attached to structure which simulates actual mounting compliance. Mechanical components of the FCS shall be duplicated dimensionally. Inertia and compliance of flight control surfaces shall be duplicated or accurately simulated. The operational mockup shall be coupled with a computer simulation of aircraft characteristics and external inputs to the flight control system. The following minimum testing shall be conducted on the operational mockup, or other appropriate test facility when approved by the procuring activity.

a. Power supply variation tests to demonstrate satisfactory operation over the range of allowable variations specified in the applicable control power specifications referenced in 3.2.5.

b. System fatigue tests (where system installation geometry or dynamic characteristics are critical to fatigue life) in accordance with MIL-A-8867 to demonstrate compliance with the requirements of 3.1.11.3. The duty cycle required shall be established by the contractor as representative of flight and ground usage.

c. Stability margin tests to verify those requirements of 3.1.3.6 which can be verified by test using an aircraft simulation or the operational mockup, but which cannot be economically or safely demonstrated in flight.

d. Tests to determine the effects of single and multiple failures on performance, safety, and mission completion reliability, and the development of emergency procedures to counteract the effects of failures.

e. Miscellaneous tests to demonstrate FCS performance, and compatibility among FCS systems and with interfacing systems.

f. System wear life 3.1.12 where component wear life is interactive.

Comparison

The C-5A flight control system testing included utilization of a functional mockup or Ironbird. The C-5A Ironbird consisted of the actual flight control hardware, control surfaces, mounting brackets with simulated main structure, and actual electrical and hydraulic power systems. Also included on the Ironbird was a pilot's station with the primary/secondary controllers and cockpit instruments. The Ironbird was electrically connected with all the C-5A automatic flight control systems hardware and to a six-degrees-of-freedom digital computer simulation of the aircraft's flight characteristics.

The following FCS tests were conducted on the C-5A Ironbird:

a. Power supply variation tests
b. Endurance
c. Frequency response/stability tests
d. System hinge moment limit compliance
e. Failure effects
f. Functional checkout including system compatibility
g. Pilot-in-the-loop final checkout and system familiarization
h. Built-in test equipment and augmentation control panel operation

Discussion

C-5A experience indicates that Ironbird FCS test results can be misleading unless careful attention is given to the structure upon which the mounting brackets are fixed. Simulating actual aircraft structural deflection in magnitude, direction and surface center of mass location is very important.

C-141 and C-5 experience indicates that piloted simulation with the Ironbird is very valuable for pilot familiarization, system development and failure effects testing. However, the C-5 program requirement for 10% endurance testing prior to first flight required 3 shift operation of the Ironbird leaving little time for piloted simulation. Early procurement of long lead items so that the Ironbird can be operational as far before first flight as possible would minimize Ironbird usage conflicts. The piloted simulation work is very helpful in debugging the new hardware thereby helping to insure first flight schedule compliance. As aircraft FCS become more complex such as with the addition of active controls for CCV the time demands on the Ironbird for piloted simulation will increase. Proper and timely incorporation of an Ironbird or Vehicle Systems Simulator into an aircraft production program can be very beneficial in lowering costs, improving safety, and producing a higher quality product.

Recommendation

Before the last sentence in the first paragraph of 4.3.2.2 add the following:

"The mockup shall include a flight station adequate for piloted simulation."

After 4.3.2.2, add Item g:

"g. Piloted simulation may be performed as a final system ground test for verification of design compliance including failure effects and for pilot familiarization."

Additional Data

Piloted flight simulation studies during development and design of the flight control systems are essential. In most cases, piloted simulation using math modelled FCS will have an important input into the FCS design and should therefore be conducted prior to the final hardware design fix date. The importance of scheduling a development program this way is becoming more apparent with the advent of Control Configured Vehicles where the impact on handling qualities in addition to system performance and aircraft safety is increasingly important.

Requiring piloted simulations prior to first flight using actual hardware will be less important in some cases depending on the flight test plans and schedules. It is essential that flight critical hardware be tested for design compliance and pilot familiarity on an Ironbird Simulator, but unless the flight test program is very short, there is no need to require piloted flight simulatiens with non-critical hardware-in-the-loop before first flight. Instead flight simulation schedules with the hardware should be dovetailed with the flight test program. Flight critical items should be tied into the Ironbird Simulator and checked out including pilot-in-the-loop testing prior to first flight. FCS such as autoland are not usually scheduled to be flight tested until well after first flight; therfore, major subsystems of that system may not be available until after the first flight date. Although simulation of that system may be required, it should be done on a schedule which is compatible with its first evaluations in flight, but should not cause delay in the first flight of the aircraft.

Requirement

4.3.2.3 Safety-of-flight Tests. Prior to first flight, sufficient testing shall be accomplished to ensure that the aircraft is safe for flight. These shall be defined in the FCS development plan and shall include, but not be limited to, the following:

4.3.2.3.1 Component Safety-of-flight Tests. All system components shall successfully demonstrate satisfactory performance and satisfactory operation under the environmental extremes expected in the flight test program. Certification that a component is safe for flight because of prior qualification and use on other aircraft may be allowed provided that the component design is identical to the previously qualified part in all significant respects and that its capability to operate under all conditions specified for its new application has been proven.

4.3.2.3.2 System Safety-of-flight Tests. The complete system shall successfully pass all of the operational mockup tests specified in 4.3.2.2 prior to first flight except that only 20 percent of the required fatigue life demonstration need be completed.

4.3.3 Aircraft Ground Tests. Prior to first flight the following minimum testing shall be performed.

a. Gain margin tests to demonstrate the zero airspeed 6 dB stability margin requirements of 3.1.3.6 for feedback systems depending on aerodynamics for loop closure and to demonstrate stability margins for nonaerodynamic loops. Primary and secondary structure shall be excited, with special attention given to areas where feedback sensors are located with loop gains increased to verify the zero airspeed requirement.

b. Functional, dynamic and static tests to demonstrate that all FCS equipment items are properly installed and that steady state responses meet FCS specification requirements. These tests shall include integrated FCS and test instrumentation as installed on the prototype airplane. Compliance with the applicable residual oscillation requirements of 3.1.3.8 shall be demonstrated.

c. Electromagnetic interference (EMI) tests to demonstrate compliance with the requirements of 3.2.5.4.1. Measurement of interference limits shall be made in accordance with MIL-STD-461 and MIL-E-6051.

d. An integrity test to insure soundness of components and connections, adequate clearances, and proper operation in accordance with MIL-A-8867.

Comparison

The C-5A aircraft and functional systems were subjected to numerous safety-of-flight quality assurance tests prior to first flight as defined by the requirements in Contract End Item (CEI) specifications CP 40002-1B, CP 40002-6B and C-5A program Category I Test Plan 3-17-1. The pre-first-flight testing was conducted, in various phases, on the aircraft, the "Ironbird" flight control system (FCS) simulator and the system components.

1. The FCS completed the following tests on the "Ironbird" prior to first flight.

    a. The FCS was subjected to 10 percent of the endurance life cycle requirements for all representative proportions of the load and stroke combinations.
    b. The FCS was tested for compliance with all the primary control characteristics such as: no objectionable characteristics, positive centering, friction, breakout forces and hysteresis, no objectionable freeplay, pilot feel forces, normal FCS travel requirements, etc.
    c. The FCS was subjected to limit load testing of the system and components.

2. The FCS components which were classed as engineering critical or procured from outside Lockheed completed the following tests prior to first flight.

    a. All components completed 10 percent of the required endurance life cycles for all representative proportions of the load and stroke combinations.
    b. All components completed static limit load testing.
    c. All components completed critical portions of MIL-STD-810 environmental testing.
    d. All components completed the normal quality assurance acceptance tests.

3. The flight test aircraft completed the following tests prior to first flight.

    a. The FCS was tested for functional, dynamic, and static compliance with all the primary control characteristics which paralleled the "Ironbird" testing noted in Paragraph 2b to ensure system integrity and compatibility.

b. The aircraft completed ground vibration testing to substantiate the flutter analysis and wind tunnel flutter model test results. These tests also confirmed the limits of the degree of stability and structural coupling with the FCS components. Residual oscillation parameters were examined to determine the safety of a first flight of the aircraft. Frequency response testing was conducted on the primary FCS through the augmentation system to insure that any non-linearities existing in the closed loop FCS did not cause any dangerous, divergent or limit cycle oscillation.

   See the validation discussions for Paragraphs 3.1.3.6 and 3.1.3.8 regarding C-5A FCS compliance regarding stability and residual oscillations.

c. Electromagnetic interference tests were conducted for compliance to MIL-STD-826 and power transients were tested for compliance to MIL-STD-704.

<u>Discussion</u>

The requirements are generally valid, demonstrable and acceptable for large transport aircraft. However, in the area of fatigue life demonstration (4.2.3.2) the C-5A procurement required 10 percent endurance cycling rather than 20 percent of required fatigue cycles prior to first flight. During the flight test development Lockheed continued FCS endurance cycling on the C-5A Ironbird and on FCS servo actuator assemblies at the vendor facilities until 100% of the endurance requirements were met or exceeded. Lockheed believes, based on both C-5A and prior Lockheed aircraft experience, that a 10 percent requirement is adequate assurance for first flight safety and subsequent flight test development which necessarily involves short flights usually over an extended time period. Lockheed believes that the 20 percent requirement of Paragraph 4.3.2.3.2 is excessive and could result in unnecessary component procurement costs and delays in airplane first flights.

<u>Recommendation</u>

Change: 4.3.2.3.2 <u>System Safety-of-Flight Tests.</u>

Replace <u>20 percent</u> with <u>10 percent</u>.

Requirement

4.3.4 Flight Tests. Flight tests shall be conducted, as defined in the FCS development plan, to demonstrate compliance with requirements where compliance cannot reasonably be demonstrated by other tests or analyses. The design and test condition guidelines tabulated in MIL-F-8785 shall be considered in establishing the flight test plan. Flight test data shall be used to verify the analytical trends predicted and shall be compared to the performance and design requirements of the FCS specification. Comparable data trends shall be required for verification where analytical data is used to extend or extrapolate flight test data to show compliance. In addition, tests shall be conducted to assure that the flight control system, in all operational states, does not violate the flutter requirements of MIL-A-8870.

Comparison

The C-5A FCS/air vehicle was subjected to flight testing to demonstrate compliance with the applicable design requirements where compliance could not otherwise be reasonably demonstrated by other tests or analysis. The flight test program was defined within a FCS development plan as noted in the validation discussion for Paragraph 4.4.1, "Flight Control System Development Plan."

The Flight Test program objectives were specified in CEI specifications, CP40002-1A and CP40002-6B, and applicable test procedures defined in Document 3-17. The applicable requirements of MIL-F-8785 were used to the extent specified in the CEI specifications. Flight test plans adhered to included the following:

- From CP40002-6B:

  "Flight test shall consist of those tests required to demonstrate the functional suitability, consistency of operation, and the accuracy of performance of the flight control subsystem/air vehicle combination for the conditions specified--The operation and performance observed or recorded shall be equal to or better than the minimum acceptable, as specified in the applicable portions of Section 3 (design performance requirements)."

- From CP40002-1A:

  "Performance. Flight test shall be conducted on a test air vehicle that is suitably configured and instrumented to obtain those data required for verification of the applicable requirements of Section 3.0 (design functional and performance requirements)."

The Flight Test program was divided into Category I and II test programs. The Category I flight testing included MFCS characteristics including ground tests (Reference MIL-F-9490D, Paragraph 4.3.3), ground handling (taxi tests),

primary flight control forces and effectiveness throughout all flight regimes, aircraft stall characteristics, minimum control speeds, aircraft operation in dives, longitudinal/lateral/directional stability and secondary flight control subsystems characteristics such as trim authority and rates and flaps, slat and ground spoiler system characteristics. AFCS functions were also evaluated during Category I flight testing.

The Category II flight test program was a joint Air Force/Lockheed effort to assure the objectives of AFR80-14, "Testing/Evaluation of Systems, Subsystems and Equipment." Tests and analysis data derived by Lockheed during Category I testing, to verify design performance requirements, were validated by selected tests during the Category II flight test program. Flutter and buzz requirements per MIL-A-8870 were flight tested for compliance to the extent noted in the validation discussions for Paragraphs 3.1.11.2, "Stiffness," and 3.2.6.7.3, "Control Surface Flutter and Buzz Prevention."

Discussion

The C-5A was analyzed for compliance with the stability and control flying quality requirements as defined in MIL-F-8785B and the results were documented in Report No. AFFDL-TR-75-3 entitled, "Evaluation of the Flying Qualities Requirements of MIL-F-8785B (ASG) using the C-5A Airplane." These validation comparisons were primarily based on Category I and II flight test results supplemented by analytical data and results obtained during the ALDCS development test program. Compliance was demonstrated to the extent noted in AFFDL-TR-75-3 and summarized by the statement, "It has been amply demonstrated through flight test and operational use that the C-5A performs its intended mission with no limitations on flight safety resulting from deficiencies in flying qualities.

"Although, as pointed out, there are quite a few areas where compliance with MIL-F-8785B (ASG) cannot be shown. Consequently, some means of deviating from those requirements would be necessary to keep contract costs within a reasonable range...."

Therefore, the flight test requirements of MIL-F-8785B, Table XV, should be used as a guideline as noted in Section 4.1, "Compliance Demonstration." "Table XV, 'Design and Test Guidelines,' gives general guidelines, but peculiarities of the specific airplane design may require additional or alternate test conditions."

On this basis the C-5A flight program has satisfied the intent of this specification. This can be demonstrated and should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Additional Data

Flight test guidelines of Table XV in Specification MIL-F-8785B are general guidelines. Test requirements for a particular aircraft may require use of additional or alternate test methods or conditions to provide compliance demonstration. Also, some areas of compliance may be more cost effectively and safely demonstrated by analysis rather than flight test. Examples where this is the case include demonstration of controllability with all engines inoperative and evaluation of controllability and vehicle stability for operations aft of the normal aft C.G. limits.

Requirement

4.4 Documentation. FCS data submittal and approval requirements for each specific model aircraft shall be in accordance with contract requirements. The data shall be furnished in accordance with appropriate line items of the Contractor Data Requirements List (DD Form 1423). Typical information and data items are listed in this section.

Comparison

The C-5A Contract End Item specification required (by supplement to the contract) that all data submittal and approval for the customer adhere to the requirements and format defined in document AZZD Exhibit 66-1. All data were furnished to the customer as specified by the appropriate line items of the contractor data requirement list (DD Form 1423).

Typical data and information items transmitted to the customer for approval and information, which were sent via this transmittal vehicle, are outlined to some extent in the validation discussions for Paragraphs 4.4.1 thru 4.4.3.3.

Discussion

The subject submittal vehicle was the instrument for periodic milestone reviews and approval of the status of the design and development phases of the C-5A FCS.

This is a valid requirement which has been satisfied by the C-5A FCS design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

4.4.1 Flight Control System Development Plan. A flight control system development plan shall be prepared by the contractor for approval by the procuring activity. This plan shall be revised and updated at intervals as specified by the procuring activity until it is mutually agreed that no further revision is required. The plan shall include a minimum of:

a. A detailed milestone chart showing the interrelationship between phases of development work to be accomplished. Design reviews shall be identified and scheduled and an outline of the progressive design verification process to be used by the contractor shall be included. Starting and completion dates for all work items and due dates for all reports shall be identified.

b. A FCS synthesis and analysis plan describing the general approach and analytical procedures to be used. Analyses planned to generate requirements for the FCS specification shall be described.

c. A verification plan defining the means selected by the contractor for verifying that the design meets each of the requirements of the FCS specification. Verification means shall be specifically correlated with each specification requirement.

d. Flight safety, reliability, maintainability, and vulnerability analysis plans to include a description of the analytical or other means selected by the contractor for design verification in these areas.

e. A functional mockup test plan, including the test procedures to be used and a listing of requirements to be satisfied by each test.

f. A ground test plan and ground test procedures defining the ground tests and functional checks to be performed prior to first flight.

g. A flight test plan and detailed flight test procedures. Each procedure shall be correlated with one or more requirements of the FCS specification.

Comparison

The C-5A aircraft was a major production aircraft development program which contractually had to be managed under the guidelines of the Contract End Item specification (CEI). The CEI specified an air vehicle development plan to establish the requirements for performance, design, test, and qualification, as well as development of the C-5A air vehicle. This is reflected in CEI specification CP 40002-1 and all related requirements therein.

The FCS subsystem development requirements are specified in CEI specification CP 40002-6B which are summarized in the "scope" as follows:

> "This volume of CP 40002 establishes the requirements for the performance, design, test and qualification of air vehicle equipment identified as the Flight Control Subsystem. The performance requirements herein specify the applicable Flight Control Subsystem requirements necessary

to accomplish the missions defined in SS 40001 and CP 40002-1."

The FCS development plan specifications were created as a result of the RFP efforts for the C-5A mission requirements and detail considerations such as functional operation, flying quality handling and performance, structural integrity, wear life, reliability, safety, maintenance, ground handling equipment, etc. This development plan established the FCS specification as outlined in the validation discussion for Paragraph 4.4.2 and the means for its implementation. The RFP included FCS specification concepts and implementation proposals which were periodically reviewed by the customer and revised as required, even after contract award, until an acceptable final configuration and development plan was established.

The FCS development plan included detailed milestone charts which indicated the interrelationship and scheduling between all phases of the design and development of the aircraft and FCS. The plan included scheduled periodic design and development progress reviews, with the customer, which was used as a means of verifying contractural compliance. The development plan and milestone charts contained a detailed outline verification process which showed progressive compliance with each of the specified CEI requirements. Starting and completion dates of all design, development, and verification work items and due dates for all periodic and final reports and documentation, were identified within the development plan and milestone charts.

A FCS synthesis and analysis plan was established as part of the main FCS development plan for the purpose of describing the general approach and analytical procedures to be used. The FCS analysis and synthesis objectives, techniques, and reports were prepared in a manner which satisfied the intent of Paragraph 4.4.3.1, "FCS Analysis Report," as noted in the validation discussion for that paragraph. The formal report describing the C-5A FCS analysis is contained in Lockheed Report No. LG1US42-2-1, Volume IV, "C-5A Flight Control Report." The FCS analysis reports and diagrams included block diagrams of the FCS. The diagrams included such parameters as transfer or describing functions, indicated the normal control paths, redundancy, location and types of sensors, and types of control devices. A general description of the FCS contained the theory of operation and modes of operation. The stability criterion and its relation to the FCS performance were described. The system characteristics were correlated with the FCS specification requirements. Where analytical predictions were sometimes used to satisfy specification requirements the assumptions, analytical approximations, and tolerances were documented and justified. Analytical data was presented for both linear, small perturbation analysis and for non-linear simulation or analysis which consider non-linearities. Where analyses techniques were used to generate requirements for the FCS specification, they were identified.

The development plan and milestone charts contained the verification means whereby Lockheed's progress and compliance was interfaced and tracked by the

procuring activity. The verification plan for all phases consisted of a description and tabulation for each of the basic design requirements of the FCS specification required by Paragraph 4.4.2. The design requirements are generally identified in the Paragraph 3.0 group, as required by the military format requirements. The design verification requirements are contained in the quality assurance sections of the Paragraph 4.0 group. As noted in the validation discussion for Paragraphs 4.1.1 and 4.3.2.1 the methods of showing compliance are divided into major subgroups of analysis, inspection, testing and demonstration. For every design requirement in Section 3.0 there is at least one verification requirement in the quality assurance section of 4.0 which requires one or more of the listed methods for showing compliance. Tabulations similar to the example shown in AFFDL-TR-74-116 "Users' Guide", Paragraph 4.4.1, Table IIID, were used as a verification completion check list for the periodic customer reviews and for documentation and data submittal requirements noted in the validation discussion for Paragraph 4.4.

This general format was used for all categories of requirements ranging from top Contract End Item specification down to the components detail specification.

The areas of flight safety, reliability and maintainability showed design verification by analysis, tests, and demonstration as specified in the quality assurance requirement section of the specification. The reliability of the FCS was validated and verified by a combination of tests and analysis. For example, tests were conducted on selected FCS components and assemblies which were significant to the air vehicle mission reliability. The analysis technique presented a mathematical reliability model of the FCS which was used to represent the contributions of various subsystem elements to successful mission accomplishment. The introduction of failure rate data into this mathematical model, was used for an analytical validation of the subsystem reliability. Some areas were verified by reliability demonstration techniques. The maintainability requirements were verified by analysis and demonstration. The objective was to achieve quantitative maintainability characteristics with a high degree of confidence. The major maintainability verification demonstration was achieved during the Category II testing.

Flight safety requirements were a part of the safety plan developed in accordance with MIL-S-38130 which was used to evaluate failure modes, malfunction effects, human error, etc. In addition to the analysis, many of the flight safety requirements were demonstrated during the Category II testing.

To supplement the Contract End Item development plan additional documents were created to expand on the detail planning. For example, Lockheed document No. 3-17 described in detail the C-5A Category I Test Plan. This covered all the quality assurance testing which included such areas as the functional mockup test plan, ground test plan, and the flight test plan. Since this represented a typical C-5A development plan instrument, a brief description is given below.

Document 3-17 was originally submitted in proposal format in response to the RFP. The submittal was revised periodically to comply with customer and contractor negotiated changes and to achieve system compatibility and the format was updated periodically to facilitate easier reference and data acquisition, until a final plan was achieved. An outline of the test objectives included:

1. the test management approach
2. laboratory testing of materials and processes
3. development and qualification of components (including Contract End Item Engineering critical components)
4. development testing of long lead time subsystems in an operational flight environment
5. flight testing of the C-5A aircraft and evaluation of system reliability, maintainability and personnel subsystems

The test plan included definitions of all planning factors such as:

1. test article descriptions
2. support aircraft required
3. special test and instrumentation requirements
4. flight test measurements to be made
5. test preparational information required
6. cross references relating test plan subparagraphs to applicable paragraphs of the design requirement documents and CEI specifications.

The basic philosophy of the test objectives was to establish compliance with the specified requirements. Other considerations were to insure that the air vehicle and its subsystems, trainers, engines, test vehicles, ground support equipment, etc., are technically sound and safe for use in Category II testing and are functionally operable, reliable, maintainable, and compatible with the specified systems. Detailed milestone charts were prepared which showed the interrelationship between the various manufacturing development, design development, and test phases. Acquisition phase testing was conducted with a uniformly controlled component development and qualification test program which was conducted at Lockheed's facility or at Lockheed's suppliers' facilities. Full scale test simulation was used early in the program to provide early identification of subsystem design changes and to insure system compatibility.

Other supplemental development plans were contained in Lockheed Documents 3-14, "Wind Tunnel Testing" and Document 3-23, "Handbook Validation."

Additional discussion is contained in the validation for Paragraphs 4.3.3, "Aircraft Ground Tests," and 4.3.4, "Flight Tests." Functional mockup and simulator tests are discussed in more detail in the validation for Paragraph 4.3.2.2.

Discussion

This is a valid requirement which has been satisfied by the C-5A control system design and can be readily demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

4.4.2 Flight Control System Specification. The contractor shall prepare a flight control system specification incorporating:

a. Applicable general system, implementation, and test requirements of this specification.

b. Special requirements of the procurement air vehicle detail specification.

c. Special requirements determined by the contractor, as required by the general specification.

A preliminary FCS specification shall be prepared within 90 days of contract award and progressively updated, as requirements are finalized.

Comparison

The C-5A FCS specification number is CP 40002-6B. It was created from a review of the then current military specification requirements in the light of C-5A mission requirements and included detailed considerations such as operability, function and performance, maintenance, AGE, safety, reliability, strength, life, environment, handling and flying qualities, and many more. The C-5A FCS concepts and the proposed implementations were reviewed and final FCS requirements were established for the performance, design, and product confirmation. Although much of this FCS specification paralleled then current MIL-F-9490 requirements, it contained significantly more detailed information which was necessary to assure the desired overall results. This document was updated as changes in the C-5A program were developed.

Discussion

The requirement is valid for future transport aircraft. It provides a worthwhile tool for the FCS development and for development monitoring by the customer. Lockheed complied with the intent of this requirement during the C-5A development. The C-5A FCS development was probably the most extensively documented of any aircraft FCS development up to that time.

Care should be taken to assure the flexibility for change is maintained to provide for an orderly development of the most cost effective FCS for the job.

Recommendation

Accept "as is."

Requirement

4.4.3 Design and Test Data Requirements. If applicable design data are available the contractor shall, in lieu of preparing new design data, use these available data supplemented by sufficient information to substantiate their applicability.

Comparison

Because the C-5A was a major production aircraft development program, the data requirements defined by 4.4.3.1 thru 4.4.3.3 were essentially satisfied. The use of existing applicable design data was used to some extent in the preparation of some portions of the subject reports.

Discussion

The context of the meaning of "existing data" may be subject to different interpretations. "Existing data" is understood to mean data which is already known from prior design, development or test experience. For example, during the preparation of the FCS analysis report per 4.4.3.1, certain of the design requirements and criteria used for the FCS analysis and synthesis may utilize work accomplished earlier in the program or data derived from other design areas.

The intent of the section 4.4.3 subparagraphs is not obvious from the present wording. Therefore, some lead in statement would be beneficial to understanding this section.

Recommendation

Clarify the requirement as follows:

> "4.4.3 Design and Test Data Requirements. The following design and test data shall be provided as required by the definitive contract requirements. If applicable design data are available the contractor shall, in lieu of preparing new design data, use these available data supplemented by sufficient information to substantiate their applicability."

Requirement

4.4.3.1 FCS Analysis Report. A report describing FCS analysis shall be prepared using an outline prepared by the contractor, subject or procuring activity approval. This report shall be initially prepared immediately following the preliminary FCS analysis and synthesis and periodically updated throughout the development period. The final update shall include as a minimum:

a. Design requirements and criteria used during the FCS analysis and synthesis.

b. Block diagrams of the FCS. These diagrams shall include transfer or describing functions and indicate normal control paths, redundancy, manual overrides, emergency provisions, location and type of sensors and control device used.

c. A general description of the FCS. The various modes of operation shall be described and the theory of operation discussed.

d. Discussions of unusual or difficult design features and problems.

e. A description of the stability and performance of the FCS and a correlation of system characteristics with the requirements of the FCS specification. Data shall be presented for both linear, small perturbation analyses and for non-linear simulations or analyses which consider nonlinearities such as actuator rate, electronic amplifier saturation, and actuator position limits. Where analytical predictions are used to satisfy specification requirements, the assumptions, analytical approximations and the tolerances placed on these analytical predictions by the contractor shall be documented and justified.

f. Results of the FCS flight safety, reliability, maintainability and vulnerability analyses. The reliability analysis results shall include a detailed listing of possible failure modes. The approach and sources of data used shall be discussed and the results compared to and correlated with requirements of the FCS specification. Analytical methods used shall be documented and justified by the contractor.

g. A general control system layout or series of layouts showing control surfaces, actuation systems, feel systems, pilot's controls and control panel organization. Means of providing redundancy and emergency provisions shall be illustrated. Layouts shall include wiring schematics for all electrical and electronic portions or the FCS and attendant electrical, hydraulic, and pneumatic power inputs to the FCS.

h. A description of piloted simulations performed, as required by 4.2.1. Where piloted simulation data is used to verify specification requirements, the simulator and flight configurations simulated shall be described and the data compared to and correlated with the requirements of the FCS specification.

i. Mathematical models of the FCS, the unaugmented airplane and other data required to allow the procuring activity to independently simulate the FCS at any point during or following the aircraft development process. Mathematical models, block diagrams, stability and performance data and layouts shall be updated following flight tests to incorporate modifications made during testing.

Comparison

The C-5A contract between Lockheed and the Air Force specified that all the items which have since been included in this requirement of MIL-F-9490D be performed by Lockheed. Lockheed's performance on each of these items was acceptable to the Air Force and approval of the final report was granted. Therefore, the C-5A is in compliance with the intent of this requirement.

Discussion

This is a good requirement in that all the appropriate items are covered.

Recommendation

Retain the requirement as stated.

Requirement

4.4.3.2 FCS Qualification and Inspection Report. The contractor shall document results of inspections used to demonstrate compliance with requirements of the FCS specification. Where inspection of component qualification status documentation is used to verify compliance with the FCS specification, the contractor prepared component specification shall be submitted as a part of the FCS inspection report.

Comparison

All C-5A FCS subsystems and components were required to meet the design compliance verification requirements under the quality assurance section 4.0. This included formal qualification requirements which specified compliance verification of the design requirements contained in the component and system specification by inspection, analysis, demonstration and/or testing.

Appropriate documentation of qualification tests was assured by the Qualification Test Procedures (QTP) which included test plans and test procedures based on the detailed test requirements and methods of performance and operation of tests. QTP's were submitted to the USAF for review and in some cases approval. The qualification testing was then accomplished based on the QTP.

C-5A qualification documentation included specification requirements and test analysis and inspection results in the final qualification reports in order to show compliance with the FCS specification requirements and to demonstrate that approved test procedures were followed.

The C-5A qualification test reports for subsystems and for engineering critical components were submitted to the USAF for approval in accordance with requirements noted in the validation discussion for Paragraph 4.4, "Documentation."

Discussion

This is a valid requirement which has been satisfied by the C-5A control system design and can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirement

4.4.3.3 FCS Test Report. A report describing and correlating tests performed and data generated to verify requirements of the FCS specification shall be prepared by the contractor. This report may be prepared in volumes, and shall include a minimum of:

a. A detailed description of the operational mockup including part numbers and the test conditions under which data was generated and a comparison of the FCS specification. Inclusion or exclusion of control surface aerodynamic hinge moments, simulation of aircraft structural compliance in lieu of airframe parts or use of other approximations in operational mockup construction shall be justified. All discrepancies or corrective actions arising from operational mockup testing shall be reported.

b. A description of the airplane ground tests performed and data generated and a discussion of any system adjustments or modifications required to satisfy requirements of the FCS specification.

c. A comparison of flight test data with requirements of the FCS specification and a description of the airplane configurations and flight conditions tested. Modifications to the FCS made during the flight test phase to meet FCS specification requirements shall be documented and justified.

Comparison

The C-5A FCS final test results reports were prepared by Lockheed and submitted for USAF approval in accordance with the documentation submittal requirements of ASZZ Exhibit 66-1 as noted in the validation discussion for Paragraph 4.4, "Documentation."

The FCS test reports followed the format of the approved test procedure plans, which assured the test requirement correlation and data generation required to verify the FCS specification requirements. This procedure is described in more detail in the validation discussion for Paragraph 4.4.1, "Flight Control System Development Plan," where Lockheed Document 3-17, "Category I Test Plan," is used to illustrate a typical development plan. Document 3-17 includes detail test and quality assurance planning and procedure requirements for all the Category I testing, which includes the operational mockup "ironbird," airplane ground tests, and flight test requirements.

The operational mockup "ironbird" for the C-5A FCS was a full scale development test simulator which was used for subsystem testing and development and "pilot in the loop" simulation. Document 3-17 and related specification documents gave a detailed description of the operational mockup including a listing of all the equipment, test requirements, and test conditions and a comparison of the FCS specification requirements. Included in the simulator testing were simulation of control surface aerodynamic hinge moments and maximum use of actual aircraft mounting structure or structural compliance simulation. The planning phase allowed for correction of discrepancies and corrective action arising from operational simulation testing. The final test report

contained a historical documentation of any discrepancies and their related corrective action or modification required to satisfy the FCS specification requirements.

Extensive C-5A airplane ground testing was conducted prior to first flight as noted in the validation discussion for Paragraph 4.3.3, "Aircraft Ground Tests." The final test report contained detailed descriptions of all tests, data generated, and compliance verification reference to the FCS requirements. The final test report contained a historical documentation of any discrepancies or adjustments and any resultant related corrective action or modification required to satisfy the FCS specification requirements.

Extensive C-5A flight testing was conducted as noted in the validation discussion for Paragraph 4.3.4. The final test report contained detail descriptions of the airplane configurations, flight test conditions, flight test requirements, data generated and compliance verification reference to the FCS requirements. The final test report contained a historical documentation of any discrepancies or adjustments, and any resultant corrective action or modification required to satisfy the FCS and air vehicle specification requirements. Any corrective action or modification documented in the final test report is justified on the basis of being required to meet the FCS specification requirements.

Discussion

This is a valid requirement which has been satisfied by the C-5A FCS and can be demonstrated. This requirement should be specified for all future transport type aircraft.

Recommendation

Retain the requirement as stated.

Requirements - Not applicable

5.0 Preparation for Delivery

5.1 Packaging Requirements

6.0 Notes

6.1 Intended Use

6.2 Procedure for Requesting Deviations

6.3 Reordered Equipment or Second Source Procurement

6.4 Users' Guide

6.5 Abbreviations

Requirement

6.6 Definitions

Recommendation

It is recommended that the following terms be added to paragraph 6.6 of MIL-F-9490D. Suggested definitions are given for those terms where a clear and consice definition was available. The determination of the definitions for the other terms must take into account the use of the term throughout MIL-F-9490D. Also, the interrelationship between terms should be included where possible in the definition, i.e., between component and module.

Aerodynamic Enhancement Flight Control System (AEFCS) See 1.2.1.2

Circuit (electrical)

Component

Computer

Electrical Computer

Function - A control function is a particular service or special duty which is performed by any portion of the FCS. Any portion of the FCS may perform more than one function.

Limiting Flight Control System (LFCS) see 1.2.1.4

Line Replacable Unit (LRU)

Mechanical Computer

Microcircuits

Module

Probable Malfunction - A probable malfunction is defined as any single electrical, hydraulic, or mechanical malfunction or failure within a utilization system which is considered probable on the basis of past service experience with similar components in aircraft applications. This is extended to multiple malfunctions when: (1) The first malfunction would not be detected during normal operation of the system, including periodic checks established at intervals which are consistent with the degree of hazard involved, or (2) The first malfunction would inevitably lead to other malfunctions.

1 sigma (σ) - Means that the probability of 68 percent of al. measured samples will fall within the spec.fied tolerance.

2 sigma (σ) - Means that the probability of 95percent of all measured samples will fall within the specified tolerance.

3 sigma (σ) - Means that the probability of 99.7 percent of all measured samples will fall within the specified tolerance.

Signal Transmission

Requirements - Not Applicable

6.7 Use of Limited Coordination Specifications

6.8 Identification of Changes

SECTION IV

CONCLUSIONS

The main objective of this general specification is to define sufficient guidelines and performance requirements so that the FCS will meet their intended functions without compromising the detail design or the interface with the aircraft and the other systems, without degrading mission performance, reliability and safety and without unduly adding to life cycle costs.

A total of 330 of the stated requirements were subjected to the validation process. Of these, it was concluded that 235 were acceptable as presented with the clarifying data contained in the "Users' Guide." Additional data for the "Users' Guide" were recommended only for clarification of 17 requirements as written. Finally, 95 requirements were recommended to be changed. These requirements changes were recommended for various reasons, some of which are listed below:

- o Too stringent for Class III aircraft
- o Too lenient or insufficient coverage
- o To provide uniformity of numerical units for related requirements
- o Requirement is beyond the scope of MIL-F-9490
- o Requirement is not clear; subject to misinterpretation
- o To improve compatibility with the intent of the specification

It was concluded that the specification represents a significant advancement towards clarifying FCS related procurement requirements. Also, the specification with the recommended revisions is suitable for transport type aircraft and leaves the contractor with sufficient design flexibility to advance the FCS state-of-the-art.

Some requirements could not be applied consistently and satisfactorily even after several attempts. Problems were experienced particularly with the application of 1.2.1 FCS Classifications and 1.2.3 FCS Criticality Classifications. A need for additional FCS classifications and a redefinition of MFCS and AFCS are believed to be necessary.

Application of this new set of requirements would probably have had only relatively minor effects on the C-5A development and final configuration because the differences in C-5A and MIL-F-9490D specifications tended to offset each other. Table 1 presents a tabular summary of the validation study. It identifies the requirements validated, indicates the requirements recommended for change, indicates the levels of C-5A compliance, and identifies requirements for which additional data was supplied for incorporation into the "Users' Guide."

TABLE 6. TABULAR SUMMARY OF C-5A VALIDATION STUDY (SYMBOLS ARE IDENTIFIED ON THE LAST SHEET OF THE TABLE)

PARAGRAPH	TITLE	SPEC. RECOMM.	LEVEL OF COMPLIANCE	STRINGENCY	TEXT FOR USER GUIDE
* 1.0	**SCOPE & CLASSIFICATIONS**				
1.1	SCOPE		F	G	
* 1.2	CLASSIFICATIONS				
* 1.2.1	FLIGHT CONTROL SYSTEM (FCS) CLASSIFICATIONS	X			
1.2.1.1	MANUAL FLIGHT CONTROL SYSTEMS (MFCS)	X	F	S	
1.2.1.2	AUTOMATIC FLIGHT CONTROL SYSTEMS (AFCS)	X	F	S	
* 1.2.2	FCS OPERATIONAL STATE CLASSIFICATIONS				
1.2.2.1	OPERATIONAL STATE I (NORMAL OPERATION)	X	F	G	
1.2.2.2	OPERATIONAL STATE II (RESTRICTED OPERATION)	X	F	G	
1.2.2.3	OPERATIONAL STATE III (MINIMUM SAFE OPERATION)		F	G	
1.2.2.4	OPERATIONAL STATE IV (CONTROLLABLE TO AN IMMEDIATE EMERGENCY LANDING)		F	G	
1.2.2.5	OPERATIONAL STATE V (CONTROLLABLE TO AN EVACUABLE FLIGHT CONDITION)		N/A	CNA	
* 1.2.3	FCS CRITICALITY CLASSIFICATIONS				
1.2.3.1	ESSENTIAL		F	G	
1.2.3.2	FLIGHT PHASE ESSENTIAL	X	F	G	
1.2.3.3	NONCRITICAL		F	G	
* 2.0	**APPLICABLE DOCUMENTS**				
2.1	(NO TITLE)	X	P	L	
2.2	OTHER PUBLICATIONS	X	P	L	
* 3.0	**REQUIREMENTS**				
3.1	SYSTEM REQUIREMENTS		F	G	
3.1.1	MFCS PERFORMANCE REQUIREMENTS		F	G	
3.1.2	AFCS PERFORMANCE REQUIREMENTS	X	F	G	
3.1.2.1	ATTITUDE HOLD (PITCH & ROLL)		F	G	
3.1.2.2	HEADING HOLD	X	P	S	X
3.1.2.3	HEADING SELECT	X	F	L	
3.1.2.4	LATERAL ACCELERATION & SIDESLIP LIMITS		F	G	
3.1.2.4.1	COORDINATION IN STEADY BANKED TURNS	X	F	L	
3.1.2.4.2	LATERAL ACCELERATION LIMITS, ROLLING		F	G	
3.1.2.4.3	COORDINATION IN STRAIGHT & LEVEL FLIGHT	X	F	L	
3.1.2.5	ALTITUDE HOLD		F	G	
3.1.2.6	MACH HOLD	X	F	L	
3.1.2.7	AIRSPEED HOLD	X	F	L	
* 3.1.2.8	AUTOMATIC NAVIGATION				
3.1.2.8.1	VOR TACAN		F	G	
3.1.2.8.1.1	VOR CAPTURE & TRACKING	X	F	L	
3.1.2.8.1.2	TACAN CAPTURE & TRACKING	X	F	L	
3.1.2.8.1.3	OVERSTATION	X	F	G	
3.1.2.9	AUTOMATIC INSTRUMENT LOW APPROACH SYSTEM	X	F	L	
3.1.2.9.1	LOCALIZER MODE	X	P	S	
3.1.2.9.2	GLIDE SLOPE MODE	X	P	S&L	
3.1.2.9.3	GO-AROUND MODE	X	P	S	
3.1.2.9.3.1	PITCH AFCS GO-AROUND		F	G	
3.1.2.9.3.2	LATERAL-HEADING AFCS GO-AROUND PERFORMANCE STANDARDS	X	P	S	

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3.1.2.9.3.3	MINIMUM GO-AROUND ALTITUDE		F	G	
3.1.2.10	ALL WEATHER LANDING SYSTEM (AWLS)	X	P	S	
3.1.2.10.1	AWLS PERFORMANCE STANDARDS – VARIATIONS OF AIRCRAFT & AIRBORNE EQUIPMENT CONFIGURATIONS	X	P	G	
3.1.2.10.2	PERFORMANCE STANDARDS – GROUND BASED EQUIPMENT VARIATIONS	X	P	S&L	
3.1.2.11	FLIGHT LOAD FATIGUE ALLEVIATION		F	L	
3.1.2.12	RIDE SMOOTHING	X	P	S	
3.1.2.12.1	RIDE DISCOMFORT INDEX		F	G	
3.1.2.13	ACTIVE FLUTTER SUPPRESSION	DNV			
3.1.2.14	GUST & MANEUVER LOAD ALLEVIATION		F	G	
3.1.2.15	AUTOMATIC TERRAIN FOLLOWING		F	G	
3.1.2.16	CONTROL STICK (OR WHEEL) STEERING	X	F	S	X
3.1.3	GENERAL FCS DESIGN		F	G	
3.1.3.1	REDUNDANCY		F	G	
3.1.3.2	FAILURE IMMUNITY & SAFETY		F	G	
3.1.3.2.1	AUTOMATIC TERRAIN FOLLOWING FAILURE IMMUNITY		F	G	X
3.1.3.3	SYSTEM OPERATION & INTERFACE		F	G	
3.1.3.3.1	WARMUP		F	G	
3.1.3.3.2	DISENGAGEMENT		F	G	X
3.1.3.3.3	MODE COMPATIBILITY		F	G	
3.1.3.3.4	FAILURE TRANSIENTS	X	F	S	
3.1.3.4	SYSTEM ARRANGEMENT		F	G	
3.1.3.5	TRIM CONTROLS	X	F	S&L	
3.1.3.6	STABILITY		F	G	
3.1.3.6.1	STABILITY MARGINS	X	F	S	
3.1.3.6.2	SENSITIVITY ANALYSIS		F	G	
3.1.3.7	OPERATION IN TURBULENCE		F	G	
3.1.3.7.1	RANDOM TURBULENCE		F	G	
3.1.3.7.2	DISCRETE GUSTS		F	G	
3.1.3.7.3	WIND MODEL FOR LANDING & TAKEOFF		F	G	
3.1.3.7.3.1	MEAN WIND		F	G	
3.1.3.7.3.2	WIND SHEAR		F	G	
3.1.3.7.3.3	WIND MODEL TURBULENCE		F	G	
3.1.3.8	RESIDUAL OSCILLATIONS	X	F	G	
3.1.3.9	SYSTEM TEST & MONITORING PROVISIONS		F	G	
3.1.3.9.1	BUILT-IN-TEST EQUIPMENT (BIT)	X	P	S	
3.1.3.9.1.1	PREFLIGHT OR PREENGAGE BIT		F	G	
3.1.3.9.1.2	MAINTENANCE BIT	X	F	L	
3.1.3.9.2	INFLIGHT MONITORING	X	P	S	
3.1.4	MFCS DESIGN		F	G	
3.1.4.1	MECHANICAL MFCS DESIGN		F	G	
3.1.4.1.1	REVERSION - BOOSTED SYSTEMS		F	G	
3.1.4.2	ELECTRICAL MFCS DESIGN	X	F	L	
3.1.4.2.1	USE OF MECHANICAL LINKAGES		F	G	
3.1.5	AFCS DESIGN		F	G	
3.1.5.1	SYSTEM REQUIREMENTS		F	G	
3.1.5.1.1	CONTROL STICK (OR WHEEL) STEERING	X	F	S	
3.1.5.1.2	FLIGHT DIRECTOR SUBSYSTEM		F	L	X
*3.1.5.2	AFCS INTERFACE				

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3.1.5 2 1	TIE-IN WITH EXTERNAL GUIDANCE		F	G	
3.1.5.2.2	SERVO ENGAGE INTERLOCKS	X	P	S	
3.1.5 2.3	ENGAGE-DISENGAGE TRANSIENTS		F	G	
* 3.1.5.3	AFCS EMERGENCY PROVISIONS				
3.1.5.3.1	MANUAL OVERRIDE CAPABILITY		F	G	
3.1.5.3.2	EMERGENCY DISENGAGEMENT		F	G	
3.1.6	MISSION ACCOMPLISHMENT RELIABILITY	X	F	L	
3.1.7	QUANTITATIVE FLIGHT SAFETY	X	F	S	
3.1.7.1	QUANTITATIVE FLIGHT SAFETY – AWLS	X	F	S	
3.1.7.1.1	ASSESSMENT OF AVERAGE RISK OF A HAZARD	X	F	L	
3.1.7.1.2	ASSESSMENT OF SPECIFIC RISK	X	P	L	
3.1.8	SURVIVABILITY		F	G	
3.1.8.1	ALL ENGINES OUT CONTROL	X	F	G	
3.1.9	INVULNERABILITY		F	G	
3.1.9.1	INVULNERABILITY TO NATURAL ENVIRONMENTS		F	G	
3.1.9.2	INVULNERABILITY TO LIGHTNING STRIKES & STATIC ATMOSPHERE ELECTRICITY		F	G	
3.1.9.3	INVULNERABILITY TO INDUCED ENVIRONMENTS		F	G	
3.1.9.4	INVULNERABILITY TO ONBOARD FAILURES OF OTHER SYSTEMS AND/OR EQUIPMENT		F	G	
3.1.9.5	INVULNERABILITY TO MAINTENANCE ERROR		F	G	
3.1.9.6	INVULNERABILITY TO PILOT & FLIGHT CREW INACTION & ERROR		F	G	
3.1.9.7	INVULNERABILITY TO ENEMY ACTION		F	G	
3.1.10	MAINTENANCE PROVISIONS		F	G	
3.1.10.1	OPERATIONAL CHECKOUT PROVISIONS		F	G	
3.1.10.2	MALFUNCTION DETECTION & FAULT ISOLATION PROVISIONS	X	P	S	
3.1.10.2 1	USE OF COCKPIT INSTRUMENTATION	X	P	S	
3.1.10 2 2	PROVISIONS FOR CHECKOUT WITH PORTABLE TEST EQUIPMENT	X	P	S	
3.1.10.3	ACCESSIBILITY & SERVICEABILITY		F	G	
3.1.10 4	MAINTENANCE PERSONNEL SAFETY PROVISIONS		F	G	
* 3.1.11	STRUCTURAL INTEGRITY				
3.1.11.1	STRENGTH		F	G	
3 1 11 1.1	DAMAGE TOLERANCE		F	G	
3 1.11.1.2	LOAD CAPABILITY OF DUAL-LOAD-PATH ELEMENTS		F	G	
3.1.11.2	STIFFNESS		F	G	
3 1.11 3	DURABILITY		F	G	X
3.1.12	WEAR LIFE		F	G	
* 3.2	SUBSYSTEM & COMPONENT DESIGN REQUIREMENTS				
3.2.1	PILOT CONTROLS & DISPLAYS	X	F	S	
3.2.1.1	PILOT CONTROLS FOR CTOL AIRCRAFT	X	P	S	
3.2.1.1.1	ADDITIONAL REQUIREMENTS FOR CONTROL STICKS	DNV			

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3.2.1.1.2	ADDITIONAL REQUIREMENT FOR RUDDER PEDALS		F	G	
3.2.1.1.3	ALTERNATE OR UNCONVENTIONAL CONTROLS		F	G	
3.2.1.1.4	VARIABLE GEOMETRY COCKPIT CONTROLS	DNV			
3.2.1.1.5	TRIM SWITCHES	X	P	S	
3.2.1.1.6	TWO-SPEED TRIM ACTUATOR	X	F	L	
3.2.1.1.7	FCS CONTROL PANEL		F	G	
3.2.1.1.8	NORMAL DISENGAGEMENT MEANS		F	G	
3.2.1.1.9	PREFLIGHT TEST CONTROLS		F	G	
3.2.1.2	PILOT CONTROLS FOR ROTARY-WING AIRCRAFT	DNV			
3.2.1.2.1	INTERCONNECTION OF COLLECTIVE PITCH CONTROL & THROTTLE(S) FOR HELICOPTERS POWERED BY RECIPROCATING ENGINE(S)	DNV			
3.2.1.2.2	INTERCONNECTION OF COLLECTIVE PITCH CONTROL & ENGINE POWER CONTROLS FOR HELICOPTERS POWERED BY TURBINE ENGINE(S)	DNV			
3.2.1.2.3	ALTERNATE OR UNCONVENTIONAL CONTROLS	DNV			
3.2.1.3	PILOT CONTROLS FOR STOL AIRCRAFT	DNV			
* 3.2.1.4	PILOT DISPLAYS				
3.2.1.4.1	FCS ANNUNCIATION		F	G	
3.2.1.4.2	FCS WARNING & STATUS ANNUNCIATION		P	G	
3.2.1.4.2.1	PREFLIGHT TEST (BIT) STATUS ANNUNCIATION		F	G	
3.2.1.4.2.2	FAILURE STATUS	X	F	G	
3.2.1.4.2.3	CONTROL AUTHORITY ANNUNCIATION	X	F	S	
3.2.1.4.3	LIFT & DRAG DEVICE POSITION INDICATORS	X	F	G	
3.2.1.4.4	TRIM INDICATORS		F	G	
3.2.1.4.5	CONTROL SURFACE POSITION INDICATION		P	G	
3.2.2	SENSORS		F	G	
* 3.2.3	SIGNAL TRANSMISSION				
* 3.2.3.1	GENERAL REQUIREMENTS				
3.2.3.1.1	CONTROL ELEMENT ROUTING		F	G	
3.2.3.1.2	SYSTEM SEPARATION, PROTECTION, & CLEARANCE		F	C	
3.2.3.1.3	FOULING PREVENTION		F	C	
3.2.3.1.4	RIGGING PROVISIONS		F	G	X
* 3.2.3.2	MECHANICAL SIGNAL TRANSMISSION				
3.2.3.2.1	LOAD CAPABILITY		F	G	
3.2.3.2.2	STRENGTH TO CLEAR OR OVERRIDE JAMMED HYDRAULIC VALVES		F	G	
3.2.3.2.3	POWER CONTROL OVERRIDE PROVISIONS		F	G	
3.2.3.2.4	CONTROL CABLE INSTALLATIONS		F	G	
3.2.3.2.4.1	CONTROL CABLE		F	G	
3.2.3.2.4.2	CABLE SIZE		F	G	
3.2.3.2.4.3	CABLE ATTACHMENTS		F	G	
3.2.3.2.4.4	CABLE ROUTING		F	G	
3.2.3.2.4.5	CABLE SHEAVES		F	G	

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3.2.3.2.4.6	CABLE & PULLEY ALIGNMENT		F	G	
3.2.3.2.4.7	PULLEY-BRACKET SPACERS	X	P	S	X
3.2.3.2.4.8	SHEAVE GUARDS		F	G	X
3.2.3.2.4.9	SHEAVE SPACING		F	G	
3.2.3.2.4.10	CABLE TENSION		F	G	
3.2.3.2.4.11	CABLE TENSION REGULATORS		F	G	
3.2.3.2.4.12	FAIRLEADS & RUBBING STRIPS		F	G	X
3.2.3.2.4.13	PRESSURE SEALS		F	G	X
3.2.3.2.5	PUSH-PULL ROD INSTALLATIONS		F	G	
3.2.3.2.5.1	PUSH-PULL ROD ASSEMBLIES	X	P	S	
3.2.3.2.5.2	LEVERS & BELLCRANKS		F	G	
3.2.3.2.5.3	PUSH-PULL ROD SUPPORTS		F	G	
3.2.3.2.5.4	PUSH-PULL ROD CLEARANCE		F	G	
3.2.3.2.6	CONTROL CHAIN		F	G	
3.2.3.2.7	PUSH-PULL FLEXIBLE CONTROLS		F	G	
3.2.3.3	ELECTRICAL SIGNAL TRANSMISSION	X	F	L	
3.2.3.3.1	ELECTRICAL FLIGHT CONTROL (EFC) INTERCONNECTIONS		F	G	
3.2.3.3.1.1	CABLE ASSEMBLY DESIGN & CONSTRUCTION		F	G	
3.2.3.3.1.2	WIRE TERMINATIONS		F	G	
3.2.3.3.1.3	INSPECTION & REPLACEMENT		F	G	
3.2.3.3.2	MULTIPLEXING		N/A	G	
*3.2.4	SIGNAL COMPUTATION				
*3.2.4.1	GENERAL REQUIREMENTS				
3.2.4.1.1	TRANSIENT POWER EFFECTS		F	G	
3.2.4.1.2	INTERCHANGEABILITY	X	F	L	
*3.2.4.1.3	COMPUTER SIGNALS				
3.2.4.1.3.1	SIGNAL TRANSMISSIONS		F	G	
3.2.4.1.3.2	SIGNAL PATH PROTECTION		F	G	
*3.2.4.2	MECHANICAL SIGNAL COMPUTATION				
3.2.4.2.1	ELEMENT LOADS		F	G	
3.2.4.2.2	GEARED MECHANISMS		F	G	
3.2.4.2.3	HYDRAULIC ELEMENTS		F	G	
3.2.4.2.4	PNEUMATIC ELEMENTS		F	G	
*3.2.4.3	ELECTRICAL SIGNAL COMPUTATION				
3.2.4.3.1	ANALOG COMPUTATION		F	C	
3.2.4.3.2	DIGITAL COMPUTATION	X	N/A	L	
3.2.4.3.2.1	MEMORY PROTECTION		N/A	G	
3.2.4.3.2.2	PROGRAM SCALING		N/A	G	
3.2.4.3.2.3	SOFTWARE SUPPORT	X	N/A	G	
*3.2.5	CONTROL POWER				
3.2.5.1	POWER CAPACITY		F	G	
3.2.5.2	PRIORITY		F	G	
3.2.5.3	HYDRAULIC POWER SUBSYSTEMS		F	G	
3.2.5.4	ELECTRICAL POWER SUBSYSTEMS		F	G	
3.2.5.4.1	ELECTROMAGNETIC INTERFERENCE LIMITS	X	P	S	
3.2.5.4.2	OVERLOAD PROTECTION		F	G	
3.2.5.4.3	PHASE SEPARATION & POLARITY REVERSAL PROTECTION	X	F	L	
3.2.5.5	PNEUMATIC POWER SUBSYSTEMS		N/A	G	
*3.2.6	ACTUATION				
*3.2.6.1	LOAD CAPABILITY				
3.2.6.1.1	LOAD CAPABILITY OF ELEMENTS SUBJECTED TO PILOT LOADS	X	F	L	X
3.2.6.1.2	LOAD CAPABILITY OF ELEMENTS DRIVEN BY POWER ACTUATORS		F	G	
3.2.6.2	MECHANICAL FORCE TRANSMITTING ACTUATION PROVISIONS	X	F	L	

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3.2.6.2.1	FORCE TRANSMITTING POWERSCREWS	X	F	L	
3.2.6.2.1.1	THREADED POWERSCREWS		F	G	
3.2.6.2.1.2	BALLSCREWS	X	F	L	
3.2.6.3	MECHANICAL TORQUE TRANSMITTING ACTUATION PROVISIONS		F	G	
3.2.6.3.1	TORQUE TUBE SYSTEMS	X	F	L	
3.2.6.3.1.1	TORQUE TUBES		F	G	
3.2.6.3.1.2	UNIVERSAL JOINTS		F	G	
3.2.6.3.1.3	SLIP JOINTS	X	F	L	
3.2.6.3.2	GEARING	X	P	L	
3.2.6.3.3	FLEXIBLE SHAFTING	DNV			
3.2.6.3.4	HELICAL SPLINES		N/A	L	
3.2.6.3.5	ROTARY MECHANICAL ACTUATORS	X	P	L	
3.2.6.3.6	TORQUE LIMITERS		F	G	
3.2.6.3.7	NO-BACK BRAKES		F	G	X
3.2.6.4	HYDRAULIC ACTUATION PROVISIONS	X	P	S	X
3.2.6.4.1	HYDRAULIC SERVOACTUATORS		F	G	X
3.2.6.4.2	MOTOR-PUMP - SERVOACTUATOR (MPS) PACKAGE	X	N/A	L	
3.2.6.4.3	ACTUATING CYLINDERS	X	F	L	X
3.2.6.4.4	FORCE SYNCHRONIZATION OF MULTIPLE HYDRAULIC SERVOACTUATORS		F	G	
3.2.6.4.5	HYDRAULIC MOTORS		F	G	
3.2.6.5	ELECTROMECHANICAL ACTUATION	X	F	L	X
3.2.6.6	PNEUMATIC ACTUATION	DNV			
3.2.6.6.1	HIGH-PRESSURE PNEUMATIC ACTUATION	DNV			
3.2.6.6.2	PNEUMATIC DRIVE TURBINES	DNV			
*3.2.6.7	INTERFACES BETWEEN ACTUATION SYSTEMS, SUPPORT STRUCTURE, & CONTROL SURFACES				
3.2.6.7.1	CONTROL SURFACE STOPS		F	G	
3.2.6.7.1.1	ADJUSTABLE STOPS		F	G	
3.2.6.7.2	CONTROL SURFACE GROUND GUST PROTECTION		F	G	
3.2.6.7.2.1	CONTROL SURFACE LOCKS	X	F	L	
3.2.6.7.2.2	PROTECTION AGAINST INFLIGHT ENGAGEMENT OF CONTROL SURFACE LOCKS	X	F	L	
3.2.6.7.3	CONTROL SURFACE FLUTTER & BUZZ PREVENTION		F	G	
*3.2.7	COMPONENT DESIGN				
*3.2.7.1	COMMON REQUIREMENTS				
3.2.7.1.1	STANDARDIZATION		F	G	
3.2.7.1.2	INTERCHANGEABILITY		F	G	
3.2.7.1.3	SELECTION OF SPECIFICATIONS & STANDARDS		F	G	
3.2.7.1.4	IDENTIFICATION OF PRODUCT		F	G	
3.2.7.1.5	INSPECTION SEALS	X	P	S	
3.2.7.1.6	MOISTURE POCKETS		F	G	
3.2.7.2	MECHANICAL COMPONENTS		F	G	
3.2.7.2.1	BEARINGS		F	S	
3.2.7.2.1.1	ANTIFRICTION BEARINGS	X	P	L	
3.2.7.2.1.2	SPHERICAL BEARINGS		F	G	
3.2.7.2.1.3	SINTERED BEARINGS		F	G	
3.2.7.2.2	CONTROLS & KNOBS		F	G	
3.2.7.2.3	DAMPERS		F	G	
3.2.7.2.4	STRUCTURAL FITTINGS		F	G	
3.2.7.2.5	LUBRICATION	X	F	S	

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3.2.7.3	ELECTRICAL & ELECTRONIC COMPONENTS	X	P	S	X
3.2.7.3.1	DIELECTRIC STRENGTH	X	F	L	
3.2.7.3.2	MICROELECTRONICS	X	F	L	
3.2.7.3.3	BURN-IN	X	F	L	X
3.2.7.3.4	SWITCHES	X	F	L	
3.2.7.3.5	THERMAL DESIGN OF ELECTRICAL & ELECTRONIC EQUIPMENT		F	G	
3.2.7.3 6	POTENTIOMETERS		F	G	
3.2.8	COMPONENT FABRICATION		F	G	
3.2.8.1	MATERIALS		F	G	
3.2.8.1.1	METALS	X	F	L	
3.2.8.1.2	NONMETALLIC MATERIALS		F	G	
3.2.8.1.3	ELECTRIC WIRE AND CABLE		F	G	
* 3.2.8.2	PROCESSES				
3.2.8.2.1	CONSTRUCTION PROCESSES	X	F	S	
3.2.8.2.2	CORROSION PROTECTION		F	G	
3.2.8.2.3	FABRICATION OF ELECTRICAL & ELECTRONIC COMPONENTS	X	F	G	
* 3.2.8.3	ASSEMBLING				
3.2.8.3.1	MECHANICAL JOINING		F	G	
3.2.8.3.1.1	JOINING WITH REMOVABLE FASTENERS		F	G	
3.2.8.3.1.2	JOINING WITH RIVETS		F	G	
3.2.8.3.1.3	THREADED JOINTS		F	G	
3.2.8.3.2	JOINT RETENTION		F	G	
3.2.8.3.2.1	RETENTION OF THREADED JOINTS		F	G	
3.2.8.3.2 2	RETENTION OF REMOVABLE FASTENERS		F	G	
3.2.8.3.2.3	USE OF RETAINER RINGS		F	G	
* 3.2.8.3.3	ASSEMBLY OF ELECTRONIC COMPONENTS				
3.2.8.3.3.1	ELECTRICAL & ELECTRONIC PART MOUNTING		F	G	
3.2.8 3.3.2	SHIELDING & BONDING OF FINISHED SURFACES		F	G	
3.2 8.3 3 3	ISOLATION OF REDUNDANT CIRCUITS		F	G	
3.2.8.3.3.4	ELECTRICAL CONNECTOR INSTALLATION		F	G	
3.2.8.3.3.5	CLEANING OF ELECTRICAL ASSEMBLIES		F	G	
* 3.2.9	COMPONENT INSTALLATION				
3.2.9.1	BASIC REQUIREMENTS		F	G	
3.2 9 2	LOCATING COMPONENTS		F	G	
3.2.9.3	INSTALLATIONS IN FUEL SYSTEM AREAS		F	G	
3.2.9.4	ELECTRICAL & ELECTRONIC COMPONENT INSTALLATIONS		F	G	
3.2.9.5	ELECTRICAL & ELECTRONIC EQUIPMENT COOLING		F	G	
3.3	ROTARY WING PERFORMANCE & DESIGN	DNV			
3.3.1	SPECIAL MFCS PERFORMANCE REQUIREMENTS	DNV			
3.3.2	SPECIAL AFCS PERFORMANCE REQUIREMENTS	DNV			
3.3.2.1	ATTITUDE HOLD (PITCH, ROLL, & YAW)	DNV			
3.3.2.2	HEADING HOLD & HEADING SELECT	DNV			
* 3 3.2 3	ALTITUDE HOLD				
3 3.2.3.1	BAROMETRIC ALTITUDE STABILIZATION	DNV			

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3.3.2.3.2	STABILIZATION OF ALTITUDE ABOVE THE TERRAIN	DNV			
3.3.2.4	HOVER HOLD	DNV			
3.3.2.5	VERNIER CONTROL FOR HOVERING	DNV			
3.3.2.6	GROUNDSPEED HOLD	DNV			
* 3.3.3	SPECIAL DESIGN REQUIREMENTS				
* 3.3.3.1	MFCS DESIGN				
3.3.3.1.1	CONTROL FEEDBACK	DNV			
3.3.3.1.2	FEEL AUGMENTATION	DNV			
3.3.3.2	AFCS DESIGN	DNV			
* 3.3.3.3	SWASHPLATE POWER ACTUATORS				
3.3.3.3.1	REDUNDANCY	DNV			
3.3.3.3.2	JAMMING	DNV			
3.3.3.3.3	FREQUENCY RESPONSE	DNV			
3.3.3.4	ACTUATION STIFFNESS	DNV			
3.3.3.5	FATIGUE LIFE DESIGN	DNV			
3.3.3.5.1	FAIL-SAFE	DNV			
3.3.3.5.2	DISPLAY	DNV			
3.3.3.6	BUILT-IN TEST	DNV			
* 4.0	QUALITY ASSURANCE				
* 4.1	GENERAL REQUIREMENTS				
4.1.1	METHODS FOR DEMONSTRATION OF COMPLIANCE		F	G	
4.1.1.1	ANALYSIS		F	G	
4.1.1.2	INSPECTION		F	G	
4.1.1.3	TEST		F	G	
4.2	ANALYSIS REQUIREMENTS		F	G	
4.2.1	PILOTED SIMULATIONS	X	F	G	
* 4.3	TEST REQUIREMENTS				
* 4.3.1	GENERAL TEST REQUIREMENTS				
4.3.1.1	TEST WITNESS	X	F	L	
4.3.1.2	ACCEPTANCE TESTS		F	G	
4.3.1.3	INSTRUMENTATION	X	F	L	
4.3.1.4	TEST CONDITIONS	X	F	L	
* 4.3.2	LABORATORY TESTS				
4.3.2.1	COMPONENT TESTS		F	G	
4.3.2.2	FUNCTIONAL MOCKUP & SIMULATOR TESTS	X	F	L	X
4.3.2.3	SAFETY-OF-FLIGHT TESTS		F	G	
4.3.2.3.1	COMPONENT SAFETY-OF-FLIGHT TESTS		F	G	
4.3.2.3.2	SYSTEM SAFETY-OF-FLIGHT TESTS	X	P	S	
4.3.3	AIRCRAFT GROUND TESTS		F	G	
4.3.4	FLIGHT TESTS		F	G	X
4.4	DOCUMENTATION		F	G	
4.4.1	FCS DEVELOPMENT PLAN		F	G	
4.4.2	FCS SPECIFICATION		F	G	
4.4.3	DESIGN & TEST DATA REQUIREMENTS	X	F	L	
4.4.3.1	FCS ANALYSIS REPORT		F	G	
4.4.3.2	FCS QUALIFICATION & INSPECTION REPORT		F	G	
4.4.3.3	FCS TEST REPORT		F	G	
* 5.0	PREPARATION FOR DELIVERY				
5.1	PACKAGING REQUIREMENTS	DNV			
* 6.0	NOTES				
6.1	INTENDED USE	DNV			
6.2	PROCEDURE FOR REQUESTING DEVIATIONS	DNV			
6.3	REORDERED EQUIPMENT OR SECOND SOURCE PROCUREMENT	DNV			

* title paragraph

PARAGRAPH	TITLE	SPEC. RECOMM.	LEVEL OF COMPLIANCE	STRINGENCY	TEXT FOR USER GUIDE
6.4	USER'S GUIDE	DNV			
6.5	ABBREVIATIONS	DNV			
6.6	DEFINITIONS	X	P	L	
6.7	USE OF LIMITED COORDINATION SPECIFICATIONS	DNV			
6.8	IDENTIFICATION OF CHANGES	DNV			

* title paragraph

Table Symbols

Specification Recommendation

(blank) - retain requirement as stated

X - recommendation made

DNV - did not validate

Level of Compliance

F - full compliance

P - partial compliance

N - no compliance

U - undetermined

N/A - not applicable to C-5A

Stringency

G - good as is

S - too strict

L - too lenient

CNA - could not assess

Text for Users Guide

(blank) - no text change

X - text provided for inclusion

SECTION V

RECOMMENDATIONS

Lockheed recommends that these proposed changes applicable to transport type aircraft be reviewed for possible incorporation into the next revision of the specification. Such considerations should take into account that detail requirements and limits for a large military transport type aircraft will, in some cases, differ from those detail requirements and limits for other types of aircraft when the aircraft are to meet the lowest life cycle costs. This is true particularly because of differences in mission types and duration, operational environment, design service life, design maneuver limits and speed limits. A comparison of these parameters for typical fighter and transport is as follows.

Parameter	Fighter	Transport
Typical Mission Duration (Hours)	1	6
Exposure to Hostile Environment	Usually	Rarely
Design Service Life (Flight Hours)	4,000	30,000 - 40,000
Design Maneuver Limits (+gs)	8-9	2.5
Speed Limits (Mach No.)	>1.0	<1.0

It is therefore recommended that where maximum cost effectiveness cannot be achieved because of unrealistic limits, detailed requirements be defined separately and realistically for various classes of aircraft. Lockheed also recommends that the additional data suitable for the "Users' Guide" and supplied in connection with the C-5A validation be incorporated into the guide to provide additional insight into the individual requirements.

Finally, Lockheed recommends that more comprehensive requirements pertinent to active controls such as for stability augmentation, lift distribution control, gust and turbulence attenuation and flutter suppression be developed for addition to the specification. Other particularly important requirements recommended for change are listed and discussed below.

1.2 Classification. FCS classifications as presently defined in the specification are a confusing mixture of control functions, system types and hardware. It has been recommended a further breakdown of classifications and new definitions related to control functions and the method for initiating control activity, but not related to system mechanization methods. "Function" has been defined in order to satisfactorily classify control systems criticality. These clarifications are believed to be important and applicable to all classes of CTOL aircraft.

1.2.3 FCS Criticality Classification. The requirements under this paragraph which define the essential criticality of FCS functions must be clarified and generally agreed upon for uniformity of interpretation. The mechanizations to achieve various flight control functions can be vastly different for a fighter and a heavy transport aircraft due to space and available power supply redundancy.

3.1.2.11 Flight Load Fatigue Alleviation. This requirement for flight load fatigue alleviation has application on the heavy transport aircraft which has operational envelopes different from the other aircraft categories. The specification is currently unclear in relating flight control and airframe structural requirements.

3.1.3.3.4 Failure Transients. The requirement dealing with failures which result in Operational State III seems to be too restrictive. Rather than specifying a maximum load factor increment (1.5 g's), structural limits along with recovery and controllability should be the major considerations. For Class III airplanes, MIL-F-8785B is more applicable.

3.1.6 Mission Accomplishment Reliability. The quantitative value stated in the requirement is believed to be unrealistic and should be revised. In addition, the requirement should be expressed in terms of the mission flight hours as recommended in this validation.

3.1.7 Quantitative Flight Safety. It is recommended that the numerical values of the aircraft loss rate specified in Table VII be revised to reflect an aircraft loss rate that is a function of mission length expressed in flight hours. In addition, the semantics of Requirement 3.1.7.1 dealing with the AWLS safety should be revised and 3.1.7.1.1 Assessment of Average Risk of a Hazard changed to 3.1.7.1.1 Hazard Risk Assessment for clarification as has been recommended in Lockheed's validation.

3.1.8 Survivability

3.1.8.1 All Engine-Out Control. Heavy transport are generally required to meet the minimum requirement of maintaining operational State IV after the loss of all engines as discussed in comment on 1.2.2.4 Operational State IV.

3.2.1.1 Pilot Controls for CTOL Aircraft. This specification should allow more design flexibility to be commensurate with the aircraft and mission requirements. This can be achieved, as has been recommended, by deletion of the last sentence of the requirement for "Strict adherence to the prescribed location and maximum range of motion of these controls is required."

REFERENCES

C-5A CONTRACT END ITEM SPECIFICATIONS (CEI)

System Specification

SS 40001B	Performance and Design Requirements for the Heavy Logistic Support System 410A, General Specification for

Prime Specification

CP 40002-1A(-1B)	Contract End Item Detail Specification, C-5A Air Vehicle-Basic (Part I) Requirements
CP 40002-2B	Contract End Item Detail Specification, C-5A Air Vehicle Air Frame Subsystems, Part I
CP 40002-5B	Contract End Item Detail Specification, C-5A Air Vehicle Secondary Power Subsystem, Part I
CP 40002-6B	Contract End Item Detail Specification, C-5A Air Vehicle Flight Control Subsystem

LOCKHEED DOCUMENTS

Document No. 3-8	C-5A Reliability Program Plan
Document No. 3-14	C-5A Wind Tunnel Testing
Document No. 3-17	C-5A Category I, Test Plan
Document No. 3-23	C-5A Handbook Validation
AZZD Exhibit 66-1	C-5A Documentation Requirements
D4M90000	General Engineering Requirements for Vendor Designed Equipment
LG1US42-2-1	C-5A Flight Control Report

OTHER PUBLICATIONS

Manuals

AFSCM80-1	Handbook of Instructions for Aircraft Design (HIAD)
FTC-TR-73-41	Category II, Evaluation of the C-5A Automatic Controls Subsystem
AFFDL-TR-75-3	Evaluation of the Flying Qualities Requirements of MIL-F-8785B (ASG) using the C-5A Airplane
ARINC Report No. 417	Design Guidelines, Air Transport Automatic Flight Control System

REFERENCES (Continued)

ARINC Characteristic No. 558	Air Transport Automatic Throttle System

Federal Aviation Authority (FAA) Documents

FAR Part 25	Federal Air Regulations - Airworthiness Standards: Transport Category Airplanes

FAA Advisory Circulars

FAA-AC-20-57A	Automatic Landing System (ALS)
FAA-AC-120-29	Criteria for Proving Category I and Category II Landing Minima for FAR 121 Operators
FAA-AC-120-28A	Criteria for Approval of Category IIIa Landing Weather Minima
FAA-AC No. 25.1329-1A	Automatic Pilot Systems Approval

FAA Technical Standard Order

FAA-TSO-C9c	Technical Standard Order, Automatic Pilots
FAA-TSO-C52a	Technical Standard Order, Flight Director

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